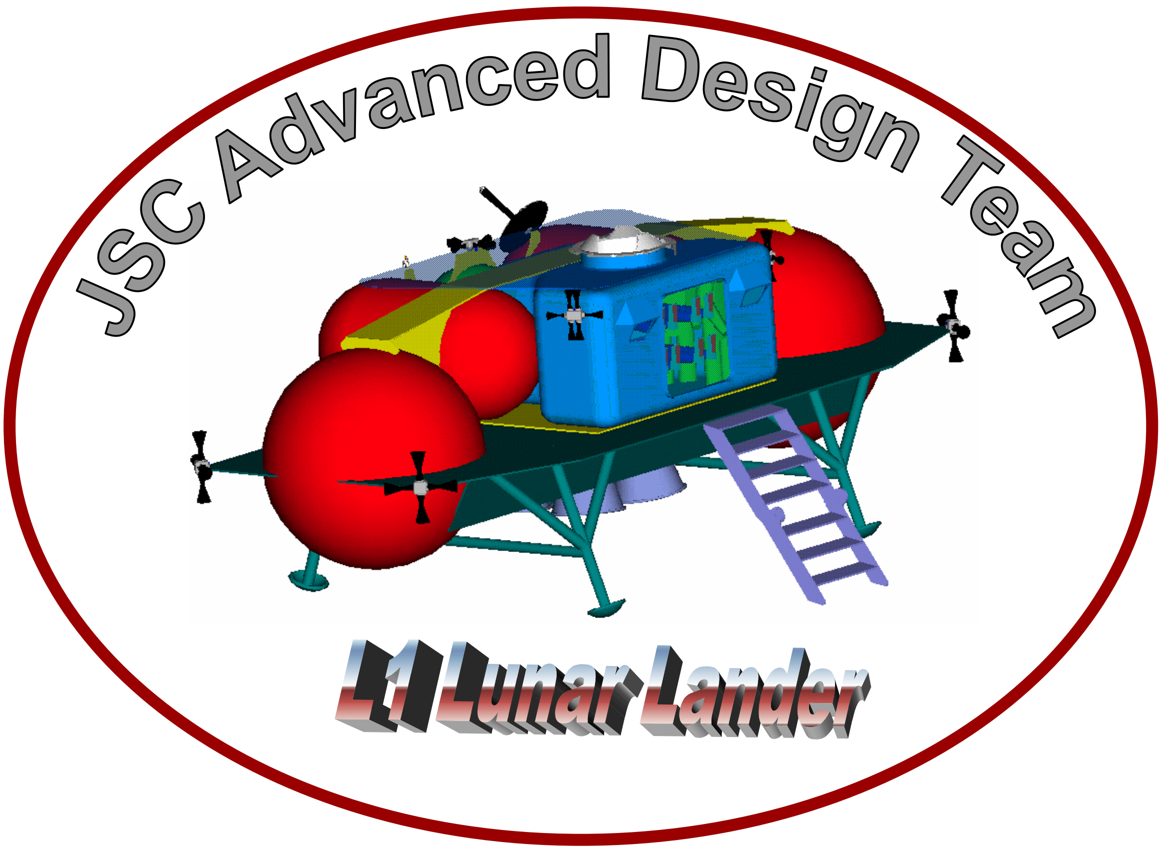
EX15-01-092

L1 Lunar Lander Element Conceptual Design Report

Engineering Directorate

November 2000



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iv. Introduction

The L1 Lunar Mission Architecture is a conceptual study which provides a possible means of returning humans to the moon within the next ten years while providing development of core capabilities that will enable human missions to Mars. Such core capabilities include the development of advanced systems and technologies that can be developed and tested in a near-earth operational environment. Such an environment will provide operational experience for autonomous deep space operations, planetary surface operations, and a Mars analog operations base at the lunar south pole.

A significant return from scientific activities may result from investigations on the lunar surface. These include a clearer understanding of the impact history of comets in near-earth space, better knowledge about the composition of the lunar mantle, past and present solar activity, lunar ice at the poles, and the history of volatiles in the solar system. Commercial potential includes the extraction of oxygen, water and metals from the lunar soil, and materials processing.

Several important assumptions are made at the outset to enable the development of the mission architecture. These include deferring the development of high-capacity launch systems by utilizing existing launch vehicle systems, and utilizing lunar libration point number one and the International Space Station (ISS) as transfer nodes between the two planetary surfaces. In addition no long-term commitment regarding extensive lunar surface infrastructure is made while initial transportation capabilities are established allowing for the future expansion of science and commercialization activities. Finally, a crew of four can be transported to and from the moon for expeditionary missions or for extended stay missions and returned to earth. Any cargo to the lunar surface is transported separately from the crew and is predeployed on the lunar surface before the crew arrives.

The L1 Lunar Mission Architecture is composed of a suite of elements which make it possible to send and return humans from the moon. These elements include a lunar depot called the Gateway which is located at L1, the Lunar Transfer Vehicle (LTV) which ferries the crew from the International Space Station to the Gateway, a high-energy injection stage which provides an initial boost for the LTV, the L1 Lunar Hab Lander which supports the crew for 30-days at the lunar south pole, the L1 Lunar Lander which performs three-day expeditionary missions to any point on the lunar surface or 30-day extended missions at the lunar south pole, and high-efficiency solar electric propulsion transfer vehicles which spiral the Gateway and landers to the L1 staging area. Other supporting elements of the architecture include the Space Shuttle which launches crew to the ISS and the Gateway to low earth orbit, the ISS which houses the LTV and serves as the nominal terminal for returning lunar astronauts, the Delta-IV expendible launch vehicle which brings the LTV and landers to low earth orbit, the Global positioning system for navigation, and an Lunar positioning system to aid in lunar navigation and communication with earth. The figure below depicts how the L1 Lunar Mission Architecture elements are deployed for to perform the mission.

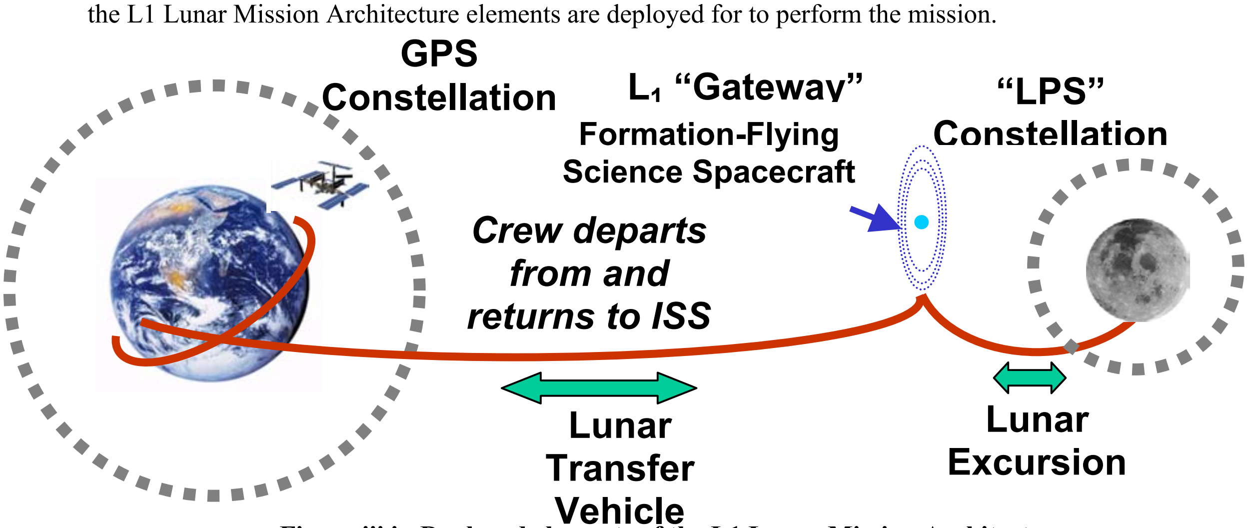


Figure iii.i: Deployed elements of the L1 Lunar Mission Architectu

L1 Lunar Lander

1.0 Element Description

The L1 Lunar Lander (L1LL) is the crew transportation element in the L1 Lunar Mission Architecture (L1LMA) which ferries the crew from the Gateway to the Lunar surface and back again. The lander is capable of supporting a crew of four for a total of nine days—three of which are spent on the lunar surface. The lander is comprised of two stages—an ascent and a descent stage. The descent stage is composed of landing gear, main propulsion system descent tanks, descent reaction control system (RCS), and support structure while the ascent stage hosts the crew module, avionics, ECLSS, ascent propulsion tanks, ascent RCS, and main propulsion system. In order to minimize the payload mass to the Gateway, the descent stage is left behind on the lunar surface. In addition to the crew, the ascent stage is capable of delivering 50 kg of Lunar samples to the Gateway for transfer back to Earth for scientific analysis.

The L1 Lunar Lander is designed to fulfill two types of missions. The first of these missions is the expeditionary-type mission where the lander is capable of sustaining a crew of four for three days at any location on the Lunar surface. In this mode the crew uses the lander as its primary base and habitation basecamp for short duration missions. The second mission for the L1 Lunar Lander is to ferry the crew to and from the L1 Lunar Hab Lander located at the Lunar south pole. In this mission the crew will live in a 30-day hab module for extended Lunar missions while the L1 Lunar Lander awaits crewed ascent in survival power mode.

Housed on the descent stage is an unpressurized rover capable of transferring the crew to and from the L1 Lunar Hab Lander (L1LHL) and which is also used as a mobility aid for crew traverses of the Lunar surface during extra-vehicular activities (EVAs). In addition to the rover, the descent stage also houses a pallet containing science payloads for use during expeditionary surface missions. Alternately, this payload pallet could be used to resupply the L1 Lunar Hab Lander.

1.1 Design Objectives, Constraints & Requirements

From a mission perspective the primary design objective is to design a Lunar lander and ascent vehicle capable of reaching any point on the Lunar surface from the Gateway at Lunar libration point one. Such global access would enable short-term expeditionary missions to explore and bring back samples from geologic sites of interest such as lava tubes, valleys, and highlands. Another mission objective is to employ spacecraft systems technologies that will have a technology readiness level (TRL) of 6 (ie. demonstration of system prototype in a relevant ground or space environment) by the year 2005. This will ensure that a proper mix of emerging and developed technologies are flown on the spacecraft in order to avoid excessive obsolescence and as a means of providing a systems technology testbed for upcoming missions to Mars. The final design objective is to identify any currently-unavailable technologies which enable this lander's mission by the 2009-2010 timeframe. This identification will help NASA managers and policy makers to decide how resources should be allocated to meet the technology development requirements for advanced human missions.

In order to minimize development costs for new launch vehicles the design team was challenged to develop a spacecraft design capable of being launched on a Boeing Delta-IV Heavy expendable launch vehicle with a projected payload mass to low-earth-orbit of 35.4 metric tons. This payload limit in turn constrained the total mass of the L1 Lunar Lander to 35.4 metric tons. The spacecraft was assumed to be fully fueled upon its arrival at the Gateway since it was not required to perform any propulsive maneuvers during its transfer aboard the solar-electric propulsion (SEP) stage. As an aid to the team the total mass of the L1 Lunar Lander was allocated amongst the different subsystems for both the ascent and descent stages. This was accomplished by using the rocket equation to determine the size of each of the L1LL stages and then allocating masses to each of the subsystems based on systems mass percentages derived from the Apollo Lunar Module (LM) mass statement. The table below lists the teams going-in mass targets at the beginning of the design exercise. It was used as a backdrop to monitor the development of the vehicle mass as vehicle system concepts were submitted for team review and approval.

L1 Lunar Lander Mass Targets				
	Ascent Stage		Descent Stage	
System	**%**	**Mass Limit**	**%**	**Mass Limit**
Structures	20%	483.20	22%	1,011.12
Propulsion	21%	507.36	27%	1,240.92
Power	15%	362.40	13%	597.48
Avionics	13%	314.08	1%	45.96
ECLSS	24%	579.84	14%	643.44
TCS	7%	169.12	7%	321.72
Science Equip.	0%	-	16%	735.36
Total Mass (kg)	100%	2,416.00	100%	4,596.00

Table 1.1: L1 Lunar Lander Dry Mass Targets

The following are the design requirements for the L1 Lunar Lander:

- 4 crew
- 8-day mission duration
  - 58-hour transit to Lunar Surface from Gateway
  - 3-day surface stay
  - 58-hour transit from Lunar surface to Gateway
- 2-stage vehicle which stages on Lunar surface
- Total delta V is 5562 m/sec (transit to/from, descent, ascent)
- Carries a 240 kg rover and a minimum of 430 kg of science equipment on the descent stage
- Ascent stage will return a minimum of 50 kg of Lunar samples to the Gateway
- Lander is capable of precision landing and hazard avoidance with manual override
- Lander is delivered to the Gateway via a Solar Electric Propulsion (SEP) stage
- Crew cabin is depressurized for surface EVAs
- Four cabin repressurizations
- 2:1 throttleable LOX/Methane main propulsion system
- LOX/Methane RCS
- Abort-to-surface (engine out), abort-to-orbit (Gateway)
- Automated rendezvous/docking w/Gateway w/ manual override
- 10.2 psi cabin atmosphere

- Package the lander within a 6 m diameter by 18 m height launch payload shroud

1.2 Vehicle Configuration Trades

Many configurations were considered for the L1 Lunar Lander, but it was quickly realized that if crew safety and utility were to be maximized that the crew egress height from the L1LL would have to be minimized. Thus landers with stages stacked one on top of the other were found to be impractical as heights above the Lunar surface reached as high as 14 m. In an effort to bring down the total vehicle height while keeping the packaging diameter within the 6 m dynamic shroud envelope several one-and-a-half stage configurations were considered. (See below.) In these configurations all the engines on the main propulsion system are used for descent and reused for ascent. On descent the engines would initially fire at full thrust and throttle back to 50% of maximum rated thrust, while on ascent the same engines would fire at full thrust from a single common bulkhead tank that was crossfed to the main engines. On ascent the main propulsion system, and crew module separate cleanly from the descent stage leaving behind descent tanks, and landing gear. In some configurations eight engines were used to generate thrust while in others only four were used. Configurations that had cylindrical or spherical shapes and configurations that had the fewest number of tanks were preferred over the others since they had the highest potential mass savings.

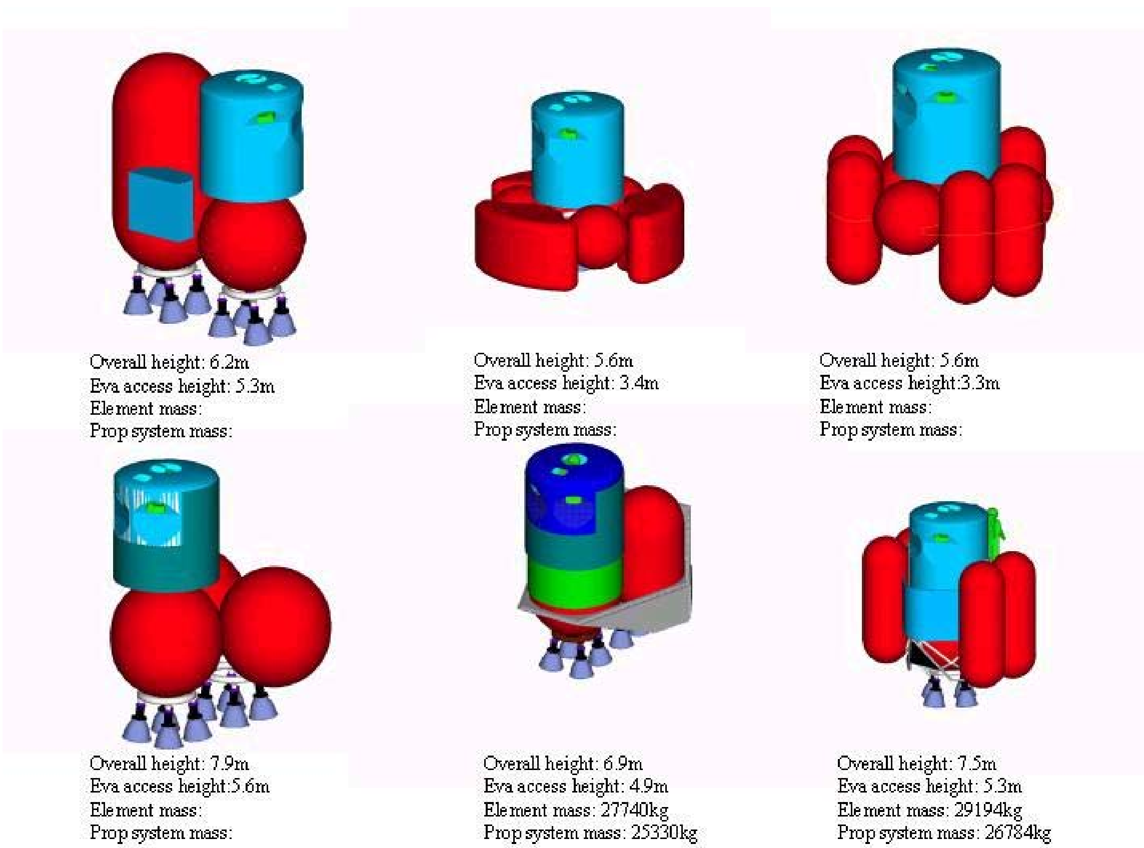


Figure 1.1: Vertical Lander Concepts

Although crew access height in these configurations had been lowered in comparison to stacked-stage landers the height remained too high. Other disadvantages to these configurations included the high numbers of engines that were envisioned to propel the landers, lateral shifts in the center of gravity caused by the depletion of descent-stage rocket propellants, no clear path for crew egress/ingress to and from the Lunar surface, and no simple strategy for deployment of thermal control system radiators.

In order to minimize the crew access height and to eliminate the issues associated with shifting center of gravity a horizontal lander concept that would be launched on-end in the payload shroud was considered. The concepts shown below are shown below. The symmetrical design simplifies the cg thrust profile while tank configurations are either spherical or cylindrical to minimize mass. The on-end launch configuration allowed the lander length to grow beyond the 6 meter launch shroud diameter so that upon landing the crew egress height could be minimized to within 2.5 meters of the Lunar surface. Furthermore, radiators for thermal control could be mounted on top of the crew cabin instead of deploying them thus reducing vehicle complexity. Due to its I-beam structure (see discussion below on structures) a platform for crew dust-off was provided so that upon L1LL ingress, Lunar dust would not be tracked into the L1 LL. As in the vertical configurations, the ascent stage with crew cabin and main propulsion system separates from the descent stage to leave behind landing gear, descent tanks, and descent structure.

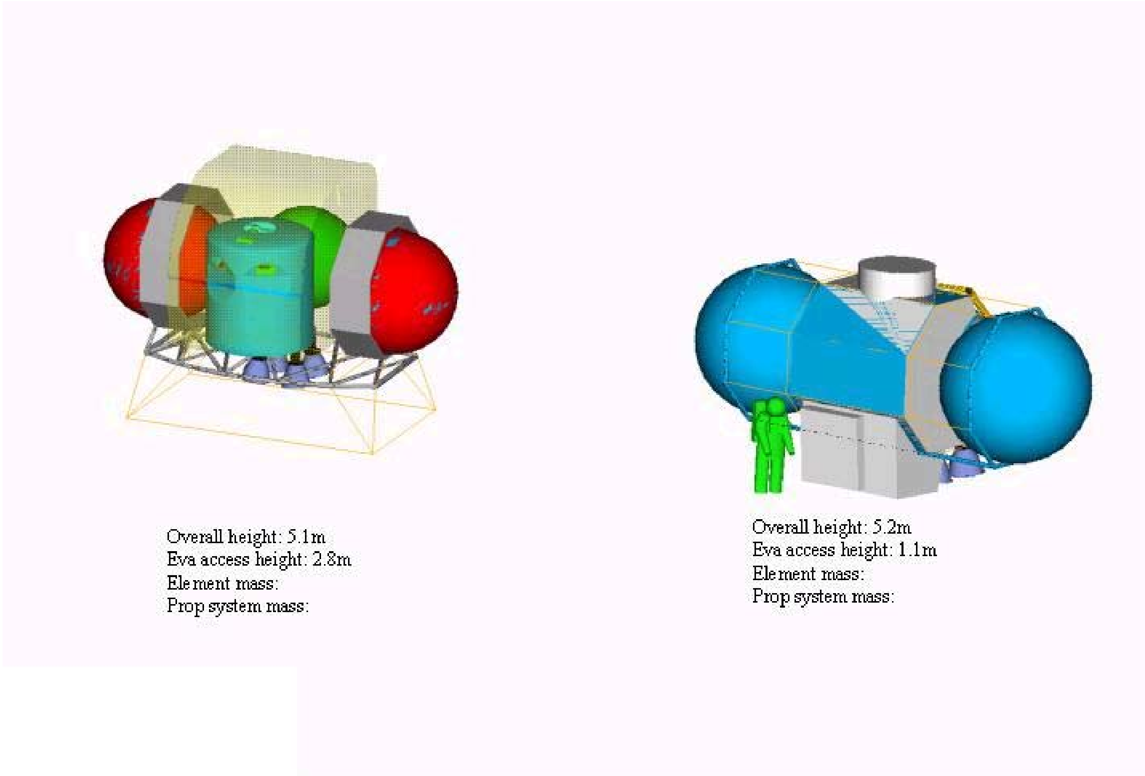


Figure 1.2: Initial Horizontal Lander Concepts

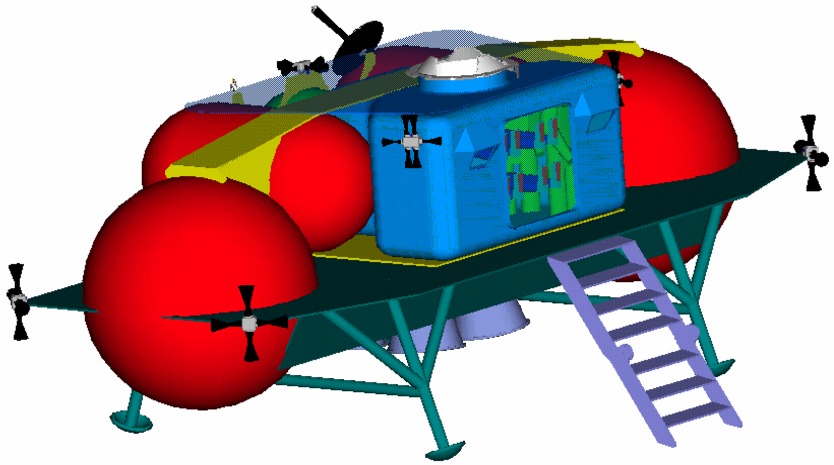


Figure 1.3: Final Configuration of the L1 Lunar Lander

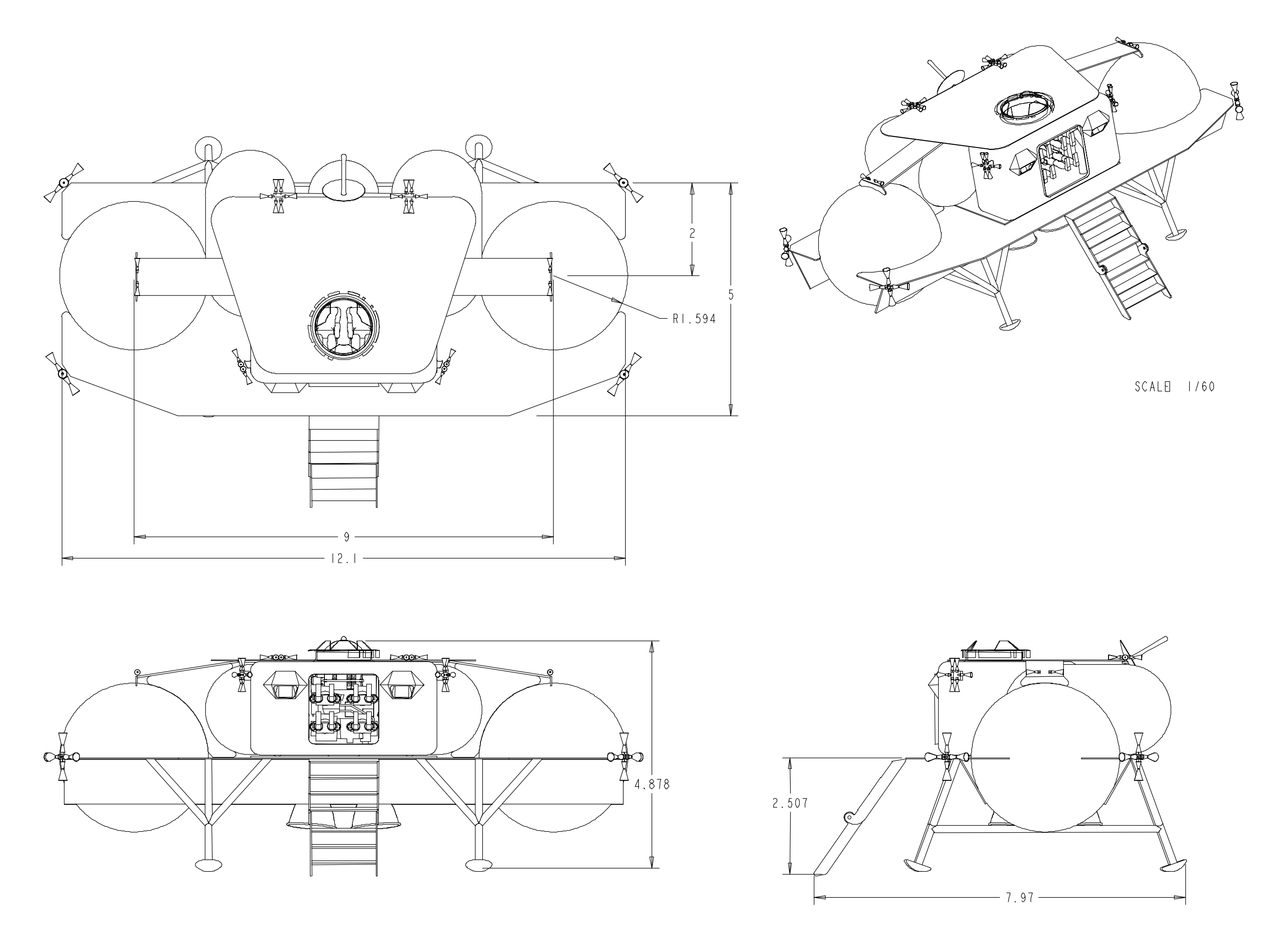


Figure 1.4: L1 Lunar Lander Four-View & Dimensions

1.3 Operations Concept

The following is a top-level operations concept for the L1 Lunar Lander:

Pre-Launch

- Lander powered up for pre-launch checkout, then placed into survival power mode for Launch

Launch

- Launch into LEO onboard Delta IV-H (unmanned)
    - Lander in survival power mode

Rendezvous with SEP

- Lander and SEP checkout (MCC)
    - Power up Lander for checkout
    - Power down Lander following checkout to survival power levels
- Rendezvous with SEP

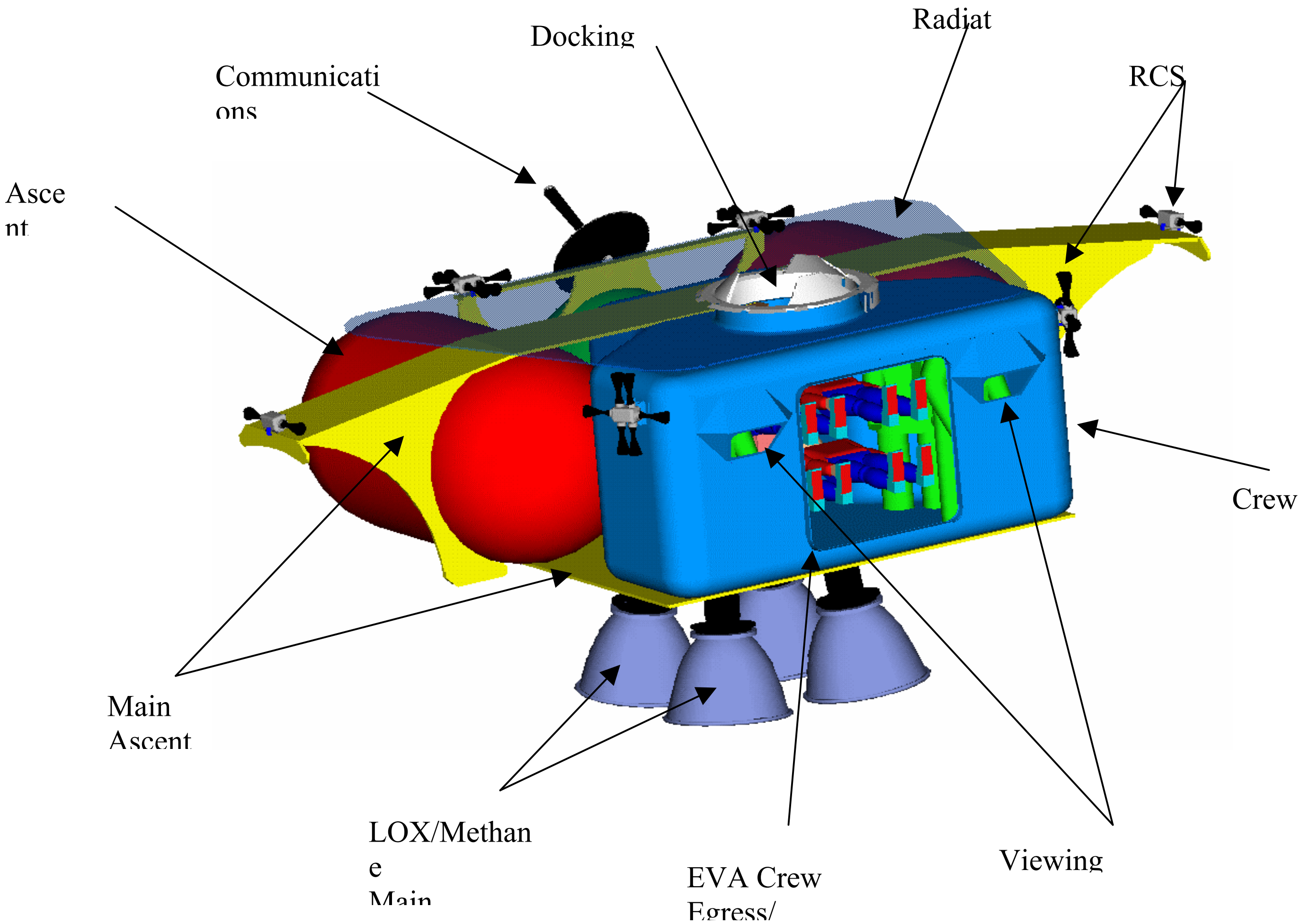


Figure 1.5: L1 Lunar Lander Ascent Stage

Operations Concept (cont.)

Dock with SEP

- Rendezvous with SEP

SEP Powered Flight

- Activate SEP; begin spiral trajectory to L1
- Monitor Lander systems during flight to L1
    - Perform avionics checkout prior to arrival at L1 (to allow time for workarounds if problems occur)

Rendezvous and Dock with Gateway

- SEP brakes Lander into L1
- Power up Lander and checkout avionics systems
- Jettison SEP
- Begin automatic Gateway rendezvous sequence
- Lander automatically rendezvous and docks with Gateway (MCC monitoring/controlling)
- Power down Lander post-docking to survival power mode levels (CC)
- Monitor Lander systems (MCC)

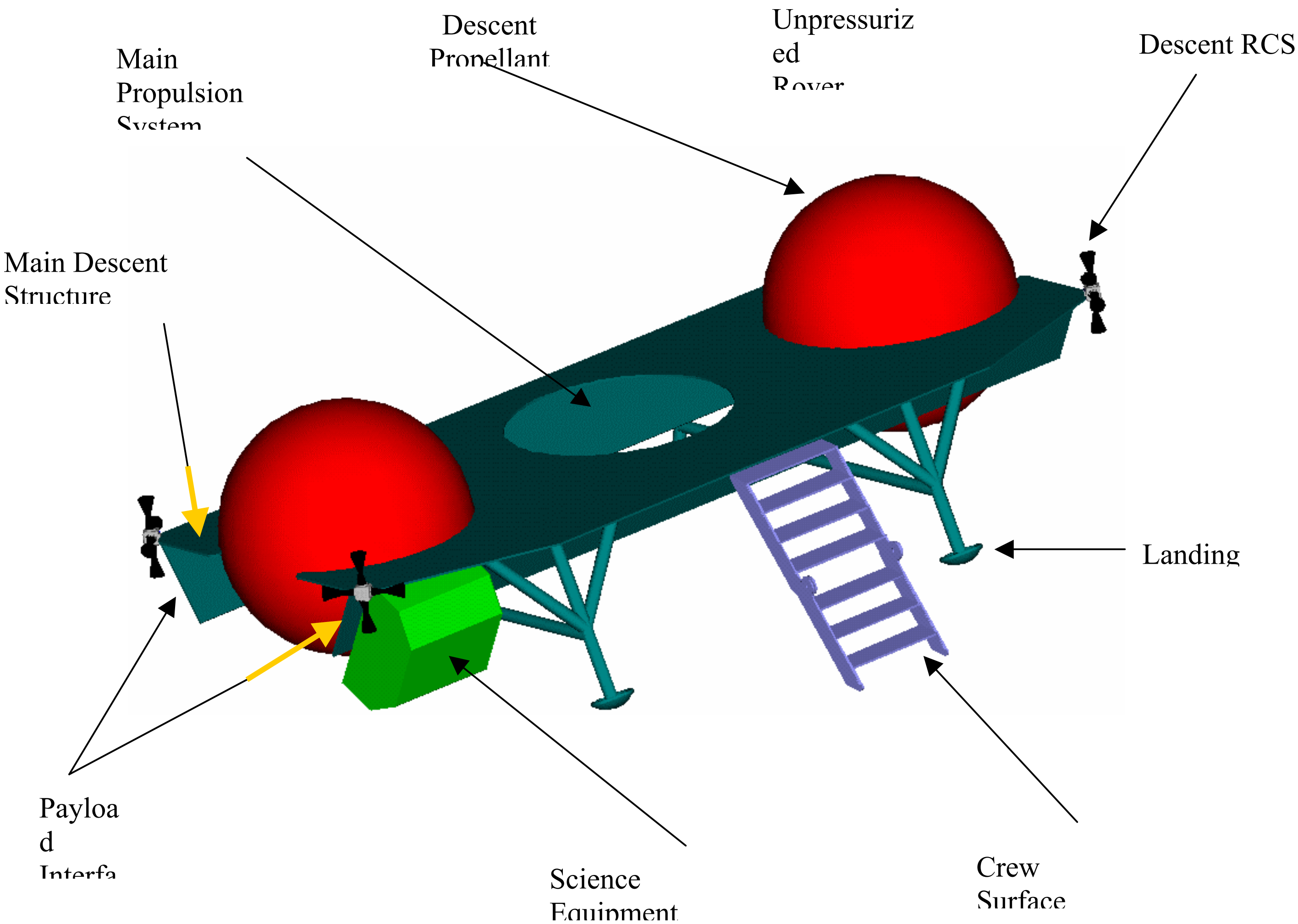


Figure 1.6: L1 Lunar Lander Descent Stage

Operations Concept (cont.)

Crew Transfer and Checkout

- Crew ingresses Lander from Gateway
- Crew powers up Lander and performs systems checkout

Coast

- Crew undocks Lander and begins Lunar descent phase
  - Gateway departure burn (234 m/s)
  - Braking burn to low Lunar orbit (634 m/s)

Powered Descent

- Descent and landing burn – Lander engines burn continuously until Lunar touchdown (1910 m/s)
  - Lander automatically performs Lunar descent – crew monitors systems and abort boundaries (crew may take over manual control if required)
  - Powered pitchover maneuver – crew monitors landing trajectory and tracks landmarks visually
  - Lander engines throttle down
  - Upon Lunar contact, Lander engines shutdown

Operations Concept (concl.)

Surface Mission

- Day 1
    - Immediately post-landing, crew and MCC perform Lander systems checks to verify that it's safe to stay on Lunar Surface for extended ops
    - Crew begins EVA prep activities
    - Lander depressurized; EVA begins
    - Crew unstows Rover and unpacks science equipment
    - Crew performs surface science/exploration activities
    - EVA ends; Lander repressurized
    - Crew meal/Crew sleep
- Day 2
    - Crew begins EVA prep activities
    - Lander depressurized; EVA begins
    - Crew unstows Rover and unpacks science equipment
    - Crew performs surface science/exploration activities
    - EVA ends; Lander repressurized
    - Crew meal/Crew sleep
- Day 3
    - Crew begins EVA prep activities
    - Lander depressurized; EVA begins
    - Crew unstows Rover and unpacks science equipment
    - Crew performs surface science/exploration activities
    - EVA ends; Lander repressurized
    - Crew meal/Crew sleep
- Day 4
    - Crew begins Lunar ascent preparations

Powered Ascent

- Lander systems checkout and countdown (crew and MCC)
- Lander automatically performs ascent burn to low Lunar orbit (1910 m/s)

Coast

- Lander automatically performs burn to depart low Lunar Orbit for Gateway (634 m/s)

Rendezvous with Gateway

- Lander performs Gateway rendezvous burn (234 m/s)

Dock with Gateway

- Lander automatically rendezvous and docks with Gateway (crew may take manual control if necessary)

Crew and Cargo Transfer

- Crew unstows Lander surface equipment and Lunar samples and transfers them to Gateway
- Lander powered down and prepared for undocking (crew and MCC)
- Lander undocked and maneuvered away from Gateway (crew and MCC)

1.4 Vehicle Mass Statement

The initial mass of the L1 Lunar Lander at the Gateway is 29,655 kg--335 kg below its original target weight of 30,000 kg and 5,735 kg below its maximum-allowed launch weight of 35,400 kg for a total growth margin potential of 16.2% While by no means conservative, the potential growth margin percentage is considered satisfactory given that the mass targets for the spacecraft (especially the ascent stage) were so tight.

L1 Lunar Lander Mass Statement				
Concept		**Total Mass**	**Wet Mass**	**Dry Mass**
Lunar Lander		29655.26	23021.18	6634.08
	Lunar Lander Electrical Power	250.00	27.00	223.00
	Lunar Field Equipment	466.38	0.00	466.38
	LL1 Space Suit	400.11	0.00	400.11
	EVA Fan	0.57	0.01	0.56
	Suit CO2 Swing Bed	4.98	0.01	4.96
	2.5 kW Thermal Control System	228.25	4.61	223.64
	1.5 kW TCS Cold Plate	14.40	0.00	0.00
	1.5 kW TCS Instruments and Controls	7.62	0.00	0.00
	1.5 kW TCS Heat Exchanger	34.45	0.00	0.00
	1.5 kW TCS MLI	90.00	0.00	0.00
	1.5 kW TCS Plumbing and Valves	24.12	1.27	22.85
	1.5 kW TCS Pumps and Accumulators		1.27	14.40
	1.5 kW TCS Radiators	45.82	1.27	44.55
	Lunar Roving Vehicle	240.61	0.00	240.61
	ECLSS + Crew Accom + Health Care	952.46	183.50	768.96
	Lunar Lander Propulsion	25800.65	22806.07	2994.58
	Lunar Lander Structure	1202.50	0.00	1202.50
	Lunar Lander Avionics	114.30	0.00	114.30

Table 1.2: L1 Lunar Lander Mass Statement

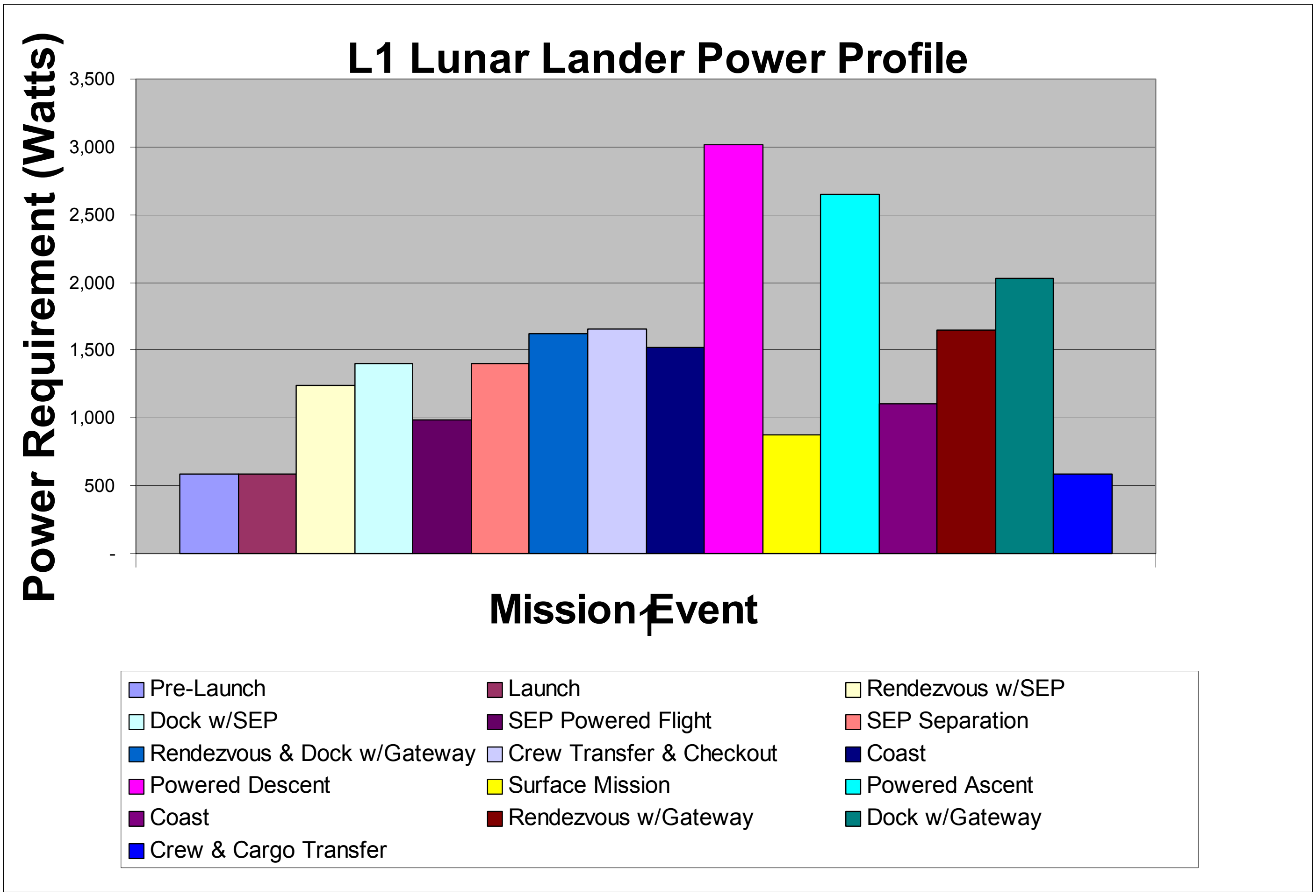
1.5 Power Profile

A power profile which details the power usage requirements for each systems component during each phase of the mission was developed. From the power profile the maximum power requirement, average power requirement and total energy consumed was generated making it possible to accurately size the power system. Listed below is the power profile summary and a chart detailing power requirements per mission phase. The power profile was used as the primary input to size the power system. It was also used to perform risk analysis.

Total Mission Without SEP Powered Flight	
Total Energy Consumed (Watt-Hours)	
[redacted]	[redacted]
[redacted] t (Watts)	
[redacted]	[redacted]
A [redacted] atts)	

Table 1.3: L1 Lunar Lander Power Requirement

Figure 1.7: Power Requirements By Mission Event



2.0 L1 Lunar Lander Descent/Ascent Propulsion System

2.1 Functional description and design requirements

The functions provided by the L1 Lunar Lander propulsion system are to provide a powered descent to the lunar surface from the Gateway. The propulsion system must be able to land the 11,593 kg wet ascent vehicle, a crew of four, and all the supplies and science needed for a three day stay. The total delta V from the Gateway to the lunar surface is 2,778 m/s.

The functions provided by the Human Ascent Vehicle propulsion system are to provide a powered ascent from the lunar surface to the Gateway. The propulsion system must be able to launch from the lunar surface, arrive and dock with the Gateway, a total payload of 2,800 kg (including the four person crew). The total delta V from the lunar surface to the Gateway is 2,778 m/s.

2.2 Trades considered and results

A parametric model for the L1 Lunar Lander with Ascent Vehicle was run considering the use of legacy storable propellants and engines (NTO/MMH & OMS) versus the more advanced but common LOx/CH4 propellants and engines. Factors considered were the higher density i.e. lower volume storage tanks and proven technology of the storable system versus higher performance, commonality (both in other vehicles and use of cryo oxygen in other on-board systems). The results of the parametric runs concluded that the LOx/CH4 propulsion system would be implemented for this design. Reasons included vehicle commonality, EVA/ECLSS/Power use of the cryogenic oxygen, active cooling of the cryogens results in lower volume and mass cryogenic tanks and higher performance. Within the choice of LOx/CH4 further trades were run on the number and shape of the cryogenic propellant tanks.

The results of the parametric runs and the final choice of LOx/CH4 engines for the L1 Lunar Lander vehicle are as follows: 4 pressure fed engines capable of 363 s ISP (5,000-lbf each), 2 spherical common bulkhead propellant tanks (OD of 3.16 m each), 24 500-lbf RCS engines with an ISP of 303 s. The dry mass of the L1 Lunar Lander propulsion system is 1,955 kg and total propellant is 16,278 kg.

The results of the parametric runs and the final choice of LOx/CH4 engines for the Human Ascent vehicle are as follows: 4 pressure fed engines (descent engines) capable of 363 s ISP (5,000-lbf each), 2 cylindrical common bulkhead propellant tanks, 24 100-lbf RCS engines with an ISP of 303 s. The dry mass of the Human Ascent propulsion system is 2,265 kg and total propellant is 6,528 kg.

2.3 Reference design description

The propulsion system chosen for the L1 Lunar Lander and Ascent vehicle was liquid oxygen liquid methane (LOx/CH4) at a mixture ratio of 3.8:1. This system is common to the Lunar Transfer Vehicle (LTV) and the Habitat Lander (HL) as well as the reference Mars missions.

The L1 Lunar Lander is powered by 4 LOx/CH4 pressure fed, 4:1 throttle ability (stored at 250 psia) engines capable of an ISP of 363 s, each with a thrust of 5,000 lbs and a length of 1.6 m. These same engines are used on the Ascent vehicle for powered ascent. The RCS on the lander consists of 24 500-lbf thrusters using the same propellants. The RCS on the ascent vehicle consists of 24 100-lbf thrusters also using LOx/CH4. The lander propulsion system was designed to land the fully loaded ascent vehicle, the crew, science and supplies needed for a three-day mission on the surface of the moon. The total delta V necessary to reach lunar orbit from the Gateway and land on the lunar surface is 2,778 m/s. The ascent vehicle was sized to provide the ascent of a 2,800 kg payload and a total delta V of 2,778 m/s from the lunar surface to the Gateway.

The propellant tank design for the L1 Lunar Lander is common to the LTV, HL, and the Mars reference mission; common bulkhead tanks storing both the oxygen and methane in their cryogenic liquid states using redundant pulse tube cryocoolers. The lander has two of these spherical common bulkhead tanks and the ascent vehicle has one. The active cooling on the lander tanks requires 354 W of input power and the ascent vehicle tank requires 128 W. The active cooling provided by the redundant cryocoolers operate at an efficiency of 18W of input power per Watt of cooling.

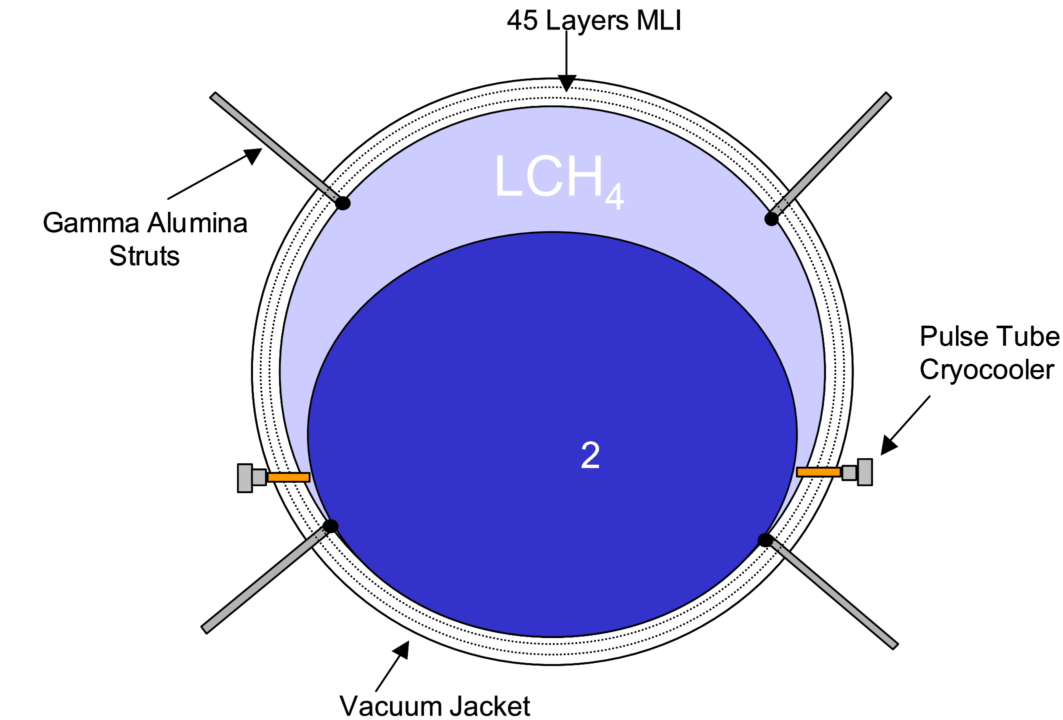


Figure 2.1: Conceptual View of a common bulkhead LOx/CH4 tank

2.4 Technology needs and design challenges

Technology needs for this design are fabrication methods for the common bulkhead cryo tanks, including interfaces for the active cooling cryocoolers, flight qualifying the 5,000-lbf LOx/CH4, 4:1 throttle capability engines, and flight qualifying lightweight pulse tube cryocoolers.

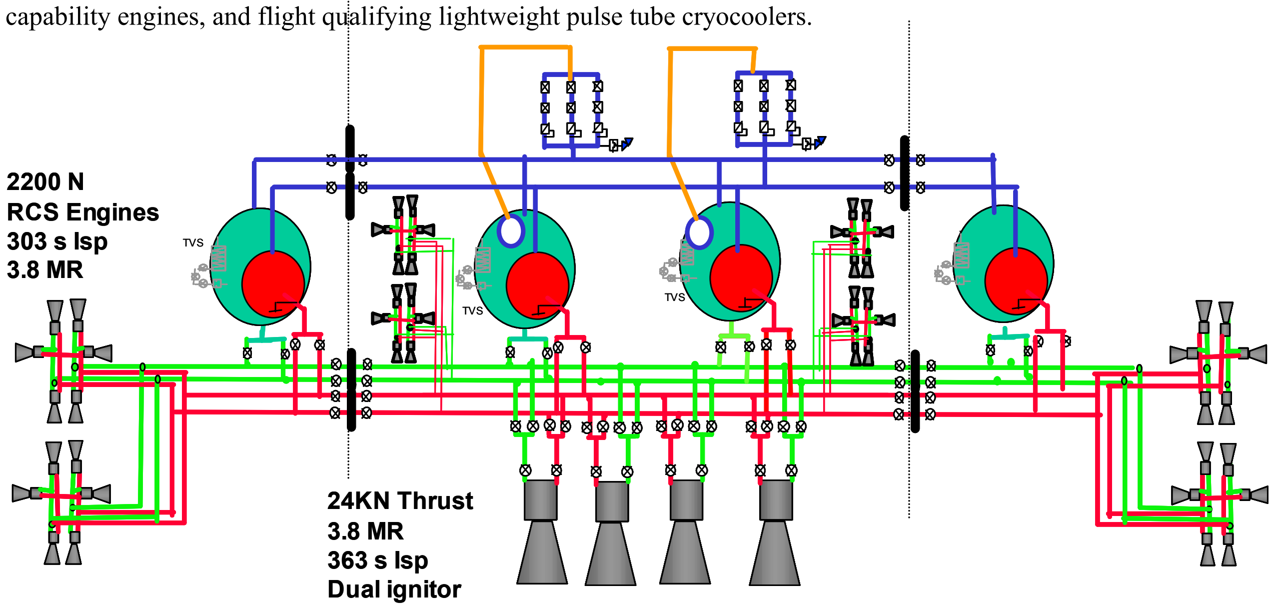


Figure 2.2: Integrated LO2/LCH4 Lander Schematic Ascent & Descent Stage

3.0 L1 Lunar Lander Structural Design

3.1 Functional description and design requirements

Maximum Payload Envelope

The maximum usable cylindrical volume within the Delta IV Heavy Launch Shroud is 18m long x 6m in diameter. This represents the static envelope of the 22.9 m long x 6.5m diameter shroud. If a dual manifest payload shroud is used the lunar lander can use an attachment location 7.81m above the base attachment point. *(Ref: 3-7 Payload Envelope, Delta IV Payload Planners Guide, Sept 1998; Delta IV Heavy Exploration Option Concept)*

Load Cases

Two load cases were considered during the design of the lunar lander structure, the load induced by launch acceleration and the load of the lander on the surface of the moon. The second load case is the limiting load case for the landing gear and the first is the limiting load case for everything else.

The launch maximum compressive acceleration loads for payloads in excess of 12,250 kg is 6.0 g's along the axis of the vehicle and 2.5 g's along the radius of the vehicle. In addition the payload must meet a minimum frequency requirement of 27 Hz to prevent coupling to the launch vehicle vibrations. *(Ref: Table 4-5 Static Envelope Requirements, Delta IV Payload Planners Guide, Sept 1998)*

The lunar surface load case is the mass of the lander without descent propellant multiplied by the lunar surface gravity. The landing gear assembly is designed to withstand the impact and leveling loads during landing by allowing honeycomb material or an open cell foam to crush on impact and absorb the load. Therefore only the static load case was used to size the landing gear.

Design Philosophy:

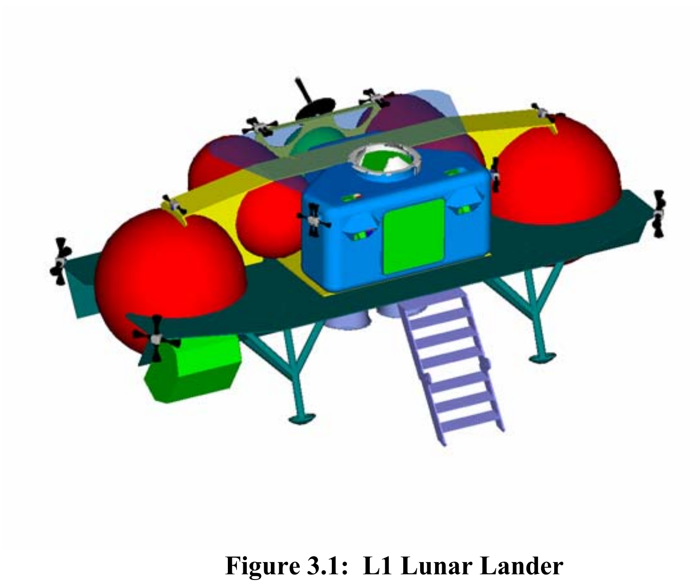


Figure 3.1: L1 Lunar Lander

The primary structure of the lunar lander is idealized into an I-beam that is supported at each end during launch. As seen in the figure above, the ascent stage structure and the descent stage structure contain the flanges of the I-beam and the tanks, tank support structure and the other components of the ascent stage comprise the web material. The integrated lander has the required structural strength and stiffness to survive launch accelerations.

The materials used in the lander structure are selected for their weight, strength, stiffness, manufacturability, and ease of analysis. Isotropic materials are used in oddly shaped structures, such as the pressure vessel and composites are used where the load paths are simple and well understood, such as the landing gear. This approach infuses a reasonable conservatism into the design that enables the design to evolve without the optimization of exotic custom composite structures.

The lander would use a dual manifest version of the Delta IV Heavy shroud, which provides interfaces for two satellites. The attachment point to the launch vehicle reside on each of the lander descent tanks. Each of the descent tanks has two bellybands which are at right angles to each other. The bellybands provide the necessary interfaces between the structure of the lunar lander and the launch vehicle. One of the bellybands is in plane with the descent stage deck to receive and transfer the load of the ascent stage during launch. The bellyband that interfaces with the launch vehicle is perpendicular to the deck and provides the upper and lower attachment points to the shroud.

The rendezvous/docking interfaces are Androgynous Peripheral Attachment Structure (APAS) common to the other architecture elements and existing hardware. No analysis was performed on the docking mechanism.

All portions of the lander structural design attempted to minimize the use of mechanisms and deployable structures to save mass.

3.2 Trades considered and results

The lander can be mounted vertically or horizontally in the launch shroud. The advantage to a vertical packaging is commonality between lunar surface load orientations and axial launch accelerations. Since the launch accelerations include a 2.5 g lateral load this benefit is superficial. The horizontal layout takes better advantage of the vehicle operational requirements and the available volume within the launch shroud.

The flanges (the large flat panels) of the ascent and descent stages were originally designed using honeycomb composite shear panels. A more detailed evaluation of the overall structural design pointed out that there is no noticeable benefit to utilizing the honeycomb material.

3.3 Reference design description

The lander structure consists of two stages connected at a single set of interfaces. The ascent stage and the descent stage are connected by the mechanisms used to separate the ascent stage during lunar liftoff.

The Descent stage structure contains two major structures; the flat panels of the lower flange of the I-beam, and the landing gear. Both structures use simple geometry to provide the required support and are constructed of graphite epoxy panels and graphite epoxy tubes respectively.

The majority of the structural complexity resides in the ascent stage. The major components of the ascent stage include: the crew pressure vessel, the APAS docking adapter, the tank attachment structure, the engine thrust structure and windows/hatches.

The internal pressure of the crew compartment is 10.2 psi. The structure is designed to support four EVAs during a three day surface mission. The crew pressure vessel is constructed of Al-Li 1095 for its high strength, and low mass. This is the same material used in the fabrication of the STS external tank pressure vessel. The shape of the pressure vessel is far from optimal from a structural standpoint so a high strength isotropic material was deemed more sensible than an exotic composite monocoque structure. In addition the pressure vessel requires an airtight liner as composite materials are known to be porous.

The APAS docking mechanism exists in a few forms (i.e.the mass of the X-38 APAS mechanism) and was thus included in the structural mass roll ups.

The tank attachment structure uses several flat composite panels to support the tanks during launch.

The engine thrust structure transfers the engine loads to the ascent stage. No specific design exists and the mass for this item is included in the vehicles 30% margin.

The miscellaneous mechanisms, hatches, and windows are allotted a weight in the overall structures concept mass rollup margin.

Structural Mass Rollup	
Ascent Stage	Mass (kg)
Upper Flange	56
Ascent Tank Thrust Structure	55
Crew Module	210
Engine Thrust Structure	32
APAS Interface	268
Descent Stage	
Lower Flange	210
Landing Gear	94
Subtotal	925
Growth Margin	30%
	1202.5

Table 3.1: Mass Rollup

3.4 Technology needs and design challenges

The lunar lander does not require the development of new materials or manufacturing techniques.

The most difficult structural element from a design and analysis perspective is the pressure vessel used to house the crew and provide support for all of the components in the vessel interior. The shape is not optimal and could benefit from additional work to verify feasibility. A large mass margin was assigned to the component to facilitate the inevitable mass growth. All of the other components have been reviewed and have been sized using conservative techniques to minimize any future mass increases. The masses of the propulsion systems tanks are included in the structures concept so they are not reflected in the margins or the mass rollups.

4.0 L1 Lunar Lander – Electrical Power System

4.1 Functional Description and Design Requirement

The Electrical Power System (EPS) will be the principal source of power necessary for the survival and for normal operation of the L1 Lunar Lander. The total Energy Consumed (watt-hours) is the sum from LEO Mission Operations Pre-Launch through Return to Gateway Crew and Cargo Transfer minus the power consumed during Transfer to Gateway SEP Powered Flight. The power requirements are:

Total Energy Consumed (watt-Hours) = 439,574
Peak Power Requirement (watts)= 3,140
Average Power Requirement (watts)= 1,312

4.2 Trades Considered and Results

As a power source two technologies were looked at; one is the Lithium primary battery technology (Li-BCX or Li-SO2), and the other is the Fuel cell. Even with an energy density of 350 to 400 Watts / kg, the Li primary cells would be much heavier than Fuel Cells. Hence, the Fuel Cell technology was chosen.

Among the Fuel Cell Technologies, Proton Exchange Membrane and Alkaline, the former was chosen because of its lesser complexity, more cycle life, less complex thermal control, and being state-of-the-art technology. A Hydrogen /Oxygen Fuel cell was chosen because Hydrocarbon (Methane) fuel would require reformer to produce H_2 from hydrocarbon and considerable technology development would thus be required in this area.

4.3 Reference Design Description

The Electrical Power System(EPS) provides power to all circuits in the L1 Lunar Lander. A H_2-O_2 PEM fuel cell would generate 28 V dc which will be distributed by a RPDA (Remote Power Distribution Assembly) to the electrical loads. The system is divided into H2-O2 storage, PEM Fuel Cell, and a Remote Power Distribution System.

This system will employ a redundancy of 3 channel (strings) sized such that 2 channels will handle peak power loads. The oxygen will be provided by the Propulsion system for the PEM fuel cell. Hydrogen will be part of the Electrical Power System. Further study will be required to determine whether liquid or high-pressure gaseous storage is more advantageous. Either system will require a cryo system to keep temperatures and hence pressures below maximum pressure.

Fuel Cell Assembly:
Power produced: 1600 w
Assuming 100w/kg, and 121 w/liter, a rough estimate of weight and volume would be
weight of 1 fuel cell = 16 kg
weight of 3 fuel cells = 48kg (total)
volume of one fuel cell=0.0132m3
and three fuel cells = 0.0396 m3 (total)

Fuel:
Weight of Hydrogen = 27 kg
Weight of Oxygen = 218 kg
Weight of water Produced 439,574 (w)/2(kwh/Kg H2O) = 220 kg H2O
Hydrogen Tank: 1.6 m3 @ 20684 kilopascals, weight will be provided by Structure Team

Propulsion will supply 218 kg of gaseous oxygen from the main propulsion system tanks thus making it unnecessary to book-keep oxygen power fuel in the electrical power system mass statement.

Electrical Power Distribution and Control:

The electrical power distribution and Control receives power from the fuel cells. A cross couple switch can switch power from one bus to the other if one of the fuel cell fails. Power is distributed by an RPCM (remote power Control Module) to the various power loads. Further down stream RPC's were not included in the power system concept.

Remote Power Controller:

3 units x 40 kg/unit = 120 kg
3units X 0.0437 m3/unit = 0.131 m3

Heat dissipation:

The H2-O2 PEM fuel cells are approximately 60% efficient.
Heat given off = ((1/0.6)-1) x 3000 peak watts = 2000 w

The EPDA is estimated 95% efficient
EPDS heat =(1-0.95) x3000 pk watts = 150 watts

Total Thermal =2000 + 150 = 2150 watts

Weight of the wire harnesses:

Assumptions: 3 wires per conductor, 20 AWG about 4m in length, 1 connector for each conductor (0.2kg each and volume 1.2cm x 1.2cm x 1.2 cm connector) .

Weight of 100 connectors @0.2kg each=20kg
Volume of 100 connector @1.2 x 1.2 x 1.2 = 1.728 cm3 = 1.728 x 10-6 m3
Weight of cable: 35 kg.

Total System:

Weight: 250 kg
Wet: 27 kg
Dry: 223 kg

Volume: 1.83m3
Pressurized: 1.60m3
Unpressurized: 0.21m3

4.4 Technology Needs and Design Challenges

- Proton Exchange Membrane (PEM) fuel cells are in the early stage of development. Limited data is available on the overall performance and reliability.

- Develop electronic switches (RPC's) to provide paralleling, programmable trip settings, higher power, and light weight.

Figures:

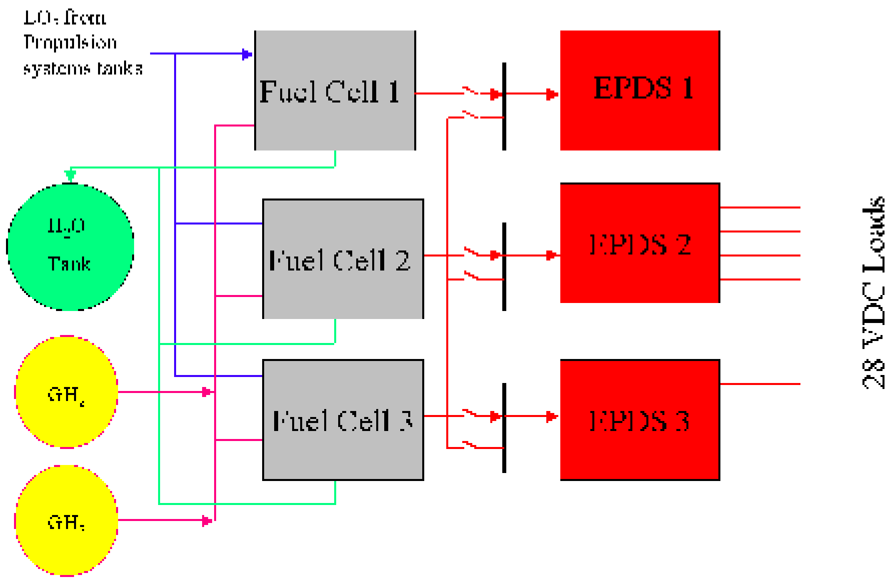


Figure 4.1: L1 Lunar Lander EPS Concept

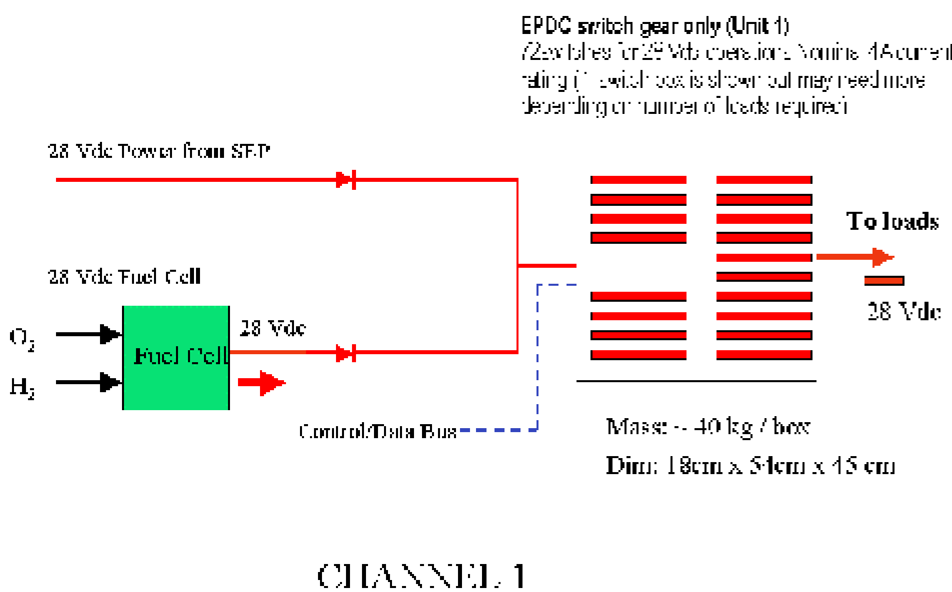


Figure 4.2: EPDC System For Lander Based On EPCU Standard H/W and Enclosures

5.0 L1 Lunar Lander Thermal Control System

5.1 Functional description and design requirements

The L1 Lunar Lander requires 2.9 kWatts peak electrical and a minimum of 0.9 kWatts during the surface stay. Adding in the metabolic and power system waste heat yields a total heat rejection required to 5.1 kWatts peak and 2.3 kWatts minimum. The L1 Lunar Lander will be required to operate for 3 days at the equator during the first third of the lunar day, which starts at sunrise. The thermal control system is considered to be a single loop ethylene-glycol/water mixture with body mounted horizontal radiators for heat rejection during coast and on-orbit operations. The ascent and descent heat loads can be handled by a sublimator which is not affected by the spacecraft orientation.

All of the electrical loads are expected to be on cold plates due to the packaging constraints and considering that the L1 Lunar Lander will be unpressurized during EVA operations. The radiator requires 12 square meters of surface area mounted on the roof of the L1 Lunar Lander.

5.2 Trades considered and results

The size and placement of the heat rejection radiators was determined by evaluating the thermal environment at 5 earth days past sunrise. Trades considered were horizontal verses vertical radiator orientations, with and without solar shades. Other options considered were a heat pump, two phase working fluid, and expendable heat rejection. These other options did not trade well due to higher mass or power requirement.

5.3 Reference design description

The thermal control system (TCS) for the L1 Lunar Lander consists of the following components:

	No Redundancy			Redundancy		
	Mass (Kg)	Power (KWATTS)	Volume (m3)	Mass (Kg)	Power (KWATTS)	Volume (m3)
heat exchangers	13.56	0.00	0.02	13.56	0.00	0.02
Cold plates	3.75	0.00	0.04	3.75	0.00	0.04
pumps with accumulators	12.00	57.50	0.04	12.00	57.50	0.04
plumbing and valves	11.15	0.00	0.00	11.15	0.00	0.00
instruments and controls	3.72	0.00	0.00	3.72	0.00	0.00
fluids	3.72	0.00	0.00	3.72	0.00	0.00
sublimator	18.98	0.00	0.04	18.98	0.00	0.04
radiators (lightweight;1-sided)	45.00	0.00	0.48	45.00	0.00	0.48
multi-layer insulation	78.00	0.00	0.51	78.00	0.00	0.51
System Total	189.88	57.5	1.13	189.88	57.5	1.13

Table 5.1: Thermal Control System Components

L1 Lunar Lander Thermal Control System Schematic

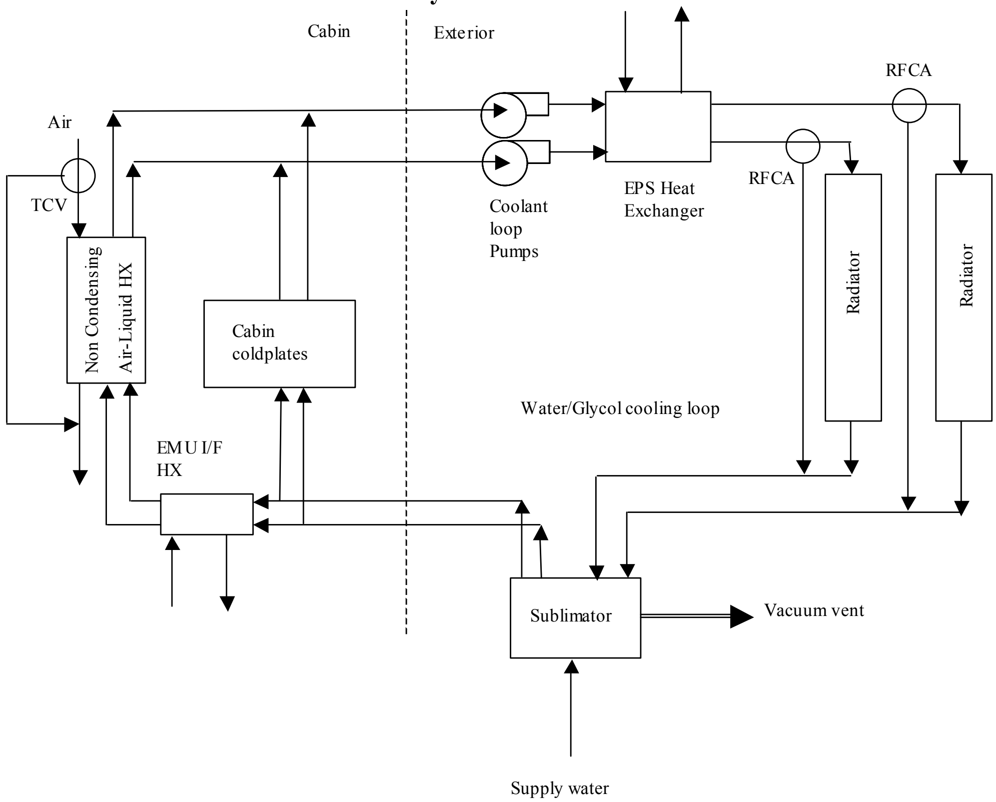


Figure 5.1: Thermal Control System Schematic

The components in the TCS are considered to be mainly shuttle type technology with the exception of the radiator. The cabin heat exchanger is non-condensing due to the ECLSS CO_2 removal subsystem which also removes moisture from the air stream. All of the cold plates are assumed to be in the crew compartment and the fluid loop pumps are redundant. The radiators are assumed to be advanced technology type with an additional radiator which is body mounted. Waste heat from the electrical power system (EPS) fuel cells is collected in a heat exchanger prior to the radiators in the external loop.

5.4 Technology needs and design challenges

The primary technology need would be the development of a lightweight radiator. An additional technology need would be to develop an integral cold plate shelf. This would save mass by reducing redundant functions between the cold plate and the shelf.

6.0 EVA System for the L1 Lunar Lander

6.1 Functional description and design requirements

The EVA system on the Lander is designed to be used for three planned, four person EVAs with the capability built into the expendable budget for one contingency, four person EVA. All are sized to be 8 hr EVAs. The system consists of the umbilical support system and the Space Suits. The Lander does not have an airlock due to tight mass budgets. The system is designed to allow the Space Suits to be worn during high-energy mission phases such as landing on the lunar surface. For this type of operation the Space Suits are attached to the vehicle ECLSS that supplies breathing air, removal of Carbon Dioxide, humidity and trace gases from the breathing loop. The vehicle also provides a heat sink so that crew person metabolic and Space Suit waste heat can be rejected. Both the ventilation loop and the water loop are driven by the prime movers in the Space Suit PLSS while connected to the ECLSS.

6.2 Trades considered and results

The Lander presented a significant challenge in EVA Space Suit recharge capability. The Space Suit oxygen system is a 3000-psi. gas-stored system which is provided via the spacecraft's main propulsion system 250-psi liquid oxygen tanks. A recharge system was conceived that includes a measured length of line that is filled with liquid oxygen. This line is shut off from the LOX supply and connected to the empty PLSS tank. Then warm water is used to provide heat to boil the LOX to 3000-psi gas thus providing a recharge for the PLSS. Such a recharge system was estimated to weigh 13.2 kg and requires the use of 0.8 kg of LOX above that needed to charge the PLSS.

6.3 Reference design description

Since the first real use of the Space Suits is from the Lander, they are first described here. The Space Suits were specifically selected to gain operational experience in preparation for Mars missions.

For this reason, the PLSS schematic chosen is the schematic that is expected to be first used on Mars. A key feature of this schematic includes CO_2 and humidity removal system that has two important features. First, to avoid having to replace an absorption canister during an EVA to accomplish a Space Suit recharge (a condition that can occur due to the need to walk back from a long distance rove using a powered rover that suffers a failure) the system is a swing bed system that is not time limited. Secondly, the system can be made to reject CO_2 from the Space Suit to the CO_2 environment of Mars (i.e. against a CO_2 partial pressure gradient). The other major technologies included in the Space Suit include a radiator toped by a membrane water boiler for heat rejection, The PLSS also provides the provision for crew person heating as well as cooling since the environments can be very cold as well as hot on the lunar surface and crewperson heating is expected to be needed on Mars which has a biased cold environment.

The Space Suit mobility garment is a back entry suit with mobility designed for surface exploration. This means the mobility is present to allow the crew person to collect rock samples, deploy scientific instruments, and move about the surface easily.

The suit and PLSS are designed to be repairable by the crew during the mission. This requires a modular architecture.

6.4 Technology needs and design challenges

Technology needs are significant for the Space Suit. The CO_2 subsystem described is currently at a TRL 3. The radiator that is small and light weight enough to be used on the PLSS and the water boiler topping unit are at TRL 3. Packaging of the PLSS in the modular arrangement needed is at TRL 2. The Space Suit garment with the mobility needed is at TRL 4. But, both PLSS and Suit are currently far too heavy for

Mars. Lunar use will be affected by the weight as well. Technology development efforts for weight reduction are currently at TRL 1 to 2. High power density, high cycle life power systems (e.g. batteries or fuel cells) are needed. Current power systems technologies needed to meet the lightweight criteria are at TRL 2-3. Another significant hole in the technology is insulation that will work in the pressurized environment of Mars. Current insulation depends on a vacuum environment. Although the moon does have a vacuum environment, the need to get operational data on the insulation layer is pressing since it is the suit layer that interacts most strongly with the dust and dirt of surface exploration. The insulation is at TRL 2. Lightweight information management systems to provide the data rates needed and provide location information are at TRL 1. Crewmember/robotic interfaces, which must be implemented as part of the information management systems, are at TRL 2-3.

Airlock system items that need technology improvement include the oxygen recharge system that is at TRL 2 and the dust management system that is at TRL 2 as well.

7.0 L1 Lunar Lander Surface Science

7.1 Functional description and design requirements

The lunar field equipment concept was developed due to the fact that astronauts once on the lunar surface will need equipment to carry out their scientific goals and objectives. The equipment will aid in a geologic survey of the landing region, as well as aid in the documentation of survey results. The equipment will also need to be fairly lightweight so that astronauts can carry it out to walk-back distances or to a nearby Lunar Habitat with ease. In addition to the lightweight basic tools and instruments, it was a self-assigned requirement that at least one larger, and more advanced item of scientific value be taken on the lander, so as to enable some technology that was not used during the Apollo era. It is also assumed that the lander will carry only equipment to acquire science data and samples and not to analyze them. That function is to be performed either on Earth or at the Lunar Habitat.

7.2 Trades Considered and Results

There were only a few trades considered in the selection of the field equipment. The first was the assumption that the lander would not carry any major scientific analysis equipment but rather tools and measuring devices. This is due to the fact that the science mass and volume budgets were significantly constrained due to the sizes of the other sub-systems. Furthermore the inside of the lander does not contain enough space to perform any kind of extended analysis. A second trade was the size of the drill. Taking the 10m drill as opposed to the 5m drill added an additional 90kg. Although that is mass taken away from other smaller equipment, it was felt that having a drill twice as long would allow the astronauts to make measurements of the lunar regolith, substantially deeper than previous attempts. This is important in particular if a return trip to the moon makes the search for water a high science priority. It was also important in making the mission operationally more similar to a Mars Mission.

7.3 Reference Design Description

The lunar field equipment consists of the following equipment:

- Drills (large, medium, & small)
- Basic field equipment (hammers, sample bags, rake, tongs, etc.)
- Camera equipment for documentation
- Geological surveying electronic equipment

These items are stored on the payload interface panels outside the cabin on the L1 Lunar Lander descent stage. They are packed in a volume efficient manner, as space on the lander is at a premium. The equipment is then offloaded upon landing. Greater care need only be taken with the removal of the 10m drill, which is significant in mass. It can be set up at the landing site, but the optimum plan calls for it to be hauled to the site of choice via an un-pressurized rover. In the overall architecture of the mission, the equipment is either put directly to use or first taken to a lunar habitat, if in proximity, that can act as a staging point for field expeditions. The overall mass is 466kg and the compacted volume is 1.3 m^3. The following tables give a more detailed look at the inventory:

Name	Unit Mass	Unit Volume	Number	Total Mass	Total Volume
	(kg)	(cm^3)		(kg)	(cm^3)
Drill string	4.0	200000	1	4.0	200000
Sample sleeve	0.1	10000	1	0.1	10000
Drill head	70.0	20000	1	70.0	20000
Mount	100.0	30000	1	100.0	30000
Sample rack	35.0	30000	1	35.0	30000
Subtotal				209.1	
25% Margin				52.3	
Total 10-M drill				**261.4**	**290000**

Table 7.1: Equipment for 10 meter Drill

Name	Unit Mass	Unit Volume	Number	Total Mass	Total Volume
	(kg)	(cm3)		(kg)	(cm3)
Electromagnetic sounder	10.0	20000	1	10.0	20000
Regolith Drill (including bits)	13.9	16704	1	13.9	16704
Rock Drill (including bits)	6.0	4000	2	12.0	8000
Geologic hammer	1.3	1200	2	2.6	2400
Chisel	0.2	100	2	0.4	200
Rake	1.5	9100	2	3.0	18200
Soil sampler	0.1	500	1	0.1	500
Small adjustable scoop	0.5	1100	2	1.0	2200
Large adjustable scoop	0.6	1200	2	1.2	2400
32-inch tongs	0.2	1600	2	0.4	3200
Long extension handle	0.5	150	2	1.0	300
4-cm drive tube	0.5	11000	45	22.5	495000
Gnomon	0.3	5300	1	0.3	5300
Orientation/Inclinometer Tool	2.0	1000	1	2.0	1000
Sample scale	0.2	900	1	0.2	900
Large tool carrier	5.9	72600	1	5.9	72600
Camera equipment	2.5	10000	3	7.5	30000
Sample collection bag	0.8	3300	100	80.0	330000
Subtotal				164.0	
25% Margin				41.0	
Total				**205.0**	**1008904**

Table 7.2: Field Equipment

Figure 7.1: Representative Core Drill, Sample Bag, Chisel

7.4 Technology Needs and Design Challenges

The primary design challenge in this case is not the selection of the equipment but rather the packaging and interface. The overall design of the lander was shown to be challenging in terms of fitting everything within the given launch shroud. Therefore the equipment will not necessarily be placed where it is most convenient, but rather where it will fit. Furthermore though most of the equipment has been tested before in space, the 10m drill will need to undergo extensive testing in a lunar-like environment, and the crew must be trained in its set-up and use. As the lander design evolves, should volume constraints become more of a problem, the drill should be scratched from the inventory and more of the other equipment should be added. In this case the drill could arrive with another landed element.

8.0 L1 Lunar Lander ECLSS

8.1 Functional Description and Design Requirements

The Environmental Control and Life Support System (ECLSS) provides the essential functions to support life and to maintain a safe, habitable environment for the crew. These functions include atmosphere control and supply, atmosphere revitalization, temperature and humidity control, fire detection and suppression, water recovery and management, food supply, and waste management. Crew accommodations and crew health care systems are also included in this section.

Top-level ECLSS design requirements and design criteria for the L1 Lunar Lander are listed below:

- 4 crew.
- 8-day crewed mission duration (58-hour transit from Gateway to lunar surface; 3-day surface stay; 58-hour transit back to Gateway).
- 100-day total mission duration including period from Earth launch to manning at Gateway (assumed for atmosphere leakage calculations).
- Crew cabin pressurized from Earth launch.
- Crew cabin atmosphere: 70.3 kPa total pressure; 70% nitrogen, 30% oxygen.
- Three, 4-person, 8-hour EVAs plus 1 contingency EVA.
- No airlock (crew cabin depressurized for EVAs).
- 4 cabin repressurizations allowed.
- 21.5 m^3 pressurized volume.
- Spacesuit umbilical support for 4 crew with oxygen and vent-loop connections. Vent-loop support provides ventilation, carbon dioxide removal, trace contaminant control, humidity control, and temperature control with the pure oxygen spacesuit atmosphere.
- Shuttle-like water allotment (3.4 kg/person/day) (mainly food rehydration and drinking water).

8.2 Trades Considered and Results

The short duration of the mission drives an open-loop system for the atmosphere gases: nitrogen and oxygen. Minimal water requirements (no shower, no laundry) coupled with the availability of fuel-cell water from the power system also eliminate the need to consider any type of water recycling. One trade study that was performed involved a comparison of non-regenerative and regenerative carbon dioxide (CO_2) removal systems. The non-regenerative system considered was a Lithium Hydroxide (LiOH) adsorbent, as used in the Apollo missions. Two regenerative CO_2 removal systems were considered, each using the same vacuum-desorbed solid-amine technology, but with different adsorbent bed size and cycle time. One system is a proposed upgrade to the Space Shuttle regenerative CO_2 removal system (RCRS). The other system is a component of an advanced spacesuit portable life support system (PLSS). This PLSS concept was selected for EVAs on the Lunar L1 mission.

Estimated masses for the three CO_2 removal options were 50 kg for LiOH, 150 kg for the RCRS upgrade (including a redundant system), and 30 kg for the PLSS system (assuming that 3 units would be required to provide nominal CO_2 removal for the 4 crew). Based on these results and commonality with the EVA hardware, the PLSS solid-amine CO_2 removal system was selected. This design allows the CO_2 removal units and any spares to be interchanged between the PLSS and cabin air revitalization systems. Further mass savings are obtained over LiOH by the capability of the solid-amine system to also remove moisture from the air and function as a humidity control system. This eliminates the need for a condensing heat exchanger and water separator in the cabin ventilation system.

8.3 Reference Design Description

The reference ECLSS design for the L1 Lunar Lander is shown schematically in Figure 8.1 and described in Table 8.1. The oxygen-enriched cabin atmosphere of 70% nitrogen and 30% oxygen minimizes EVA pre-breathe requirements while maintaining materials selection within the range currently tested and approved for space flight. (By comparison, the Apollo Lunar Module used a pure oxygen atmosphere.) This mixed cabin atmosphere, coupled with the need to depressurize the entire cabin during EVAs (no

airlock), and the need to provide umbilical support to the suited crew were primary drivers in the ECLSS design.
Cryogenic oxygen is used in the design to provide spacesuit purge and umbilical support as well as to provide for cabin atmosphere make-up due to crew metabolic usage and leakage. A common source of cryogenic oxygen for propulsion, power, and ECLSS is assumed (including ascent-stage and descent-stage tanks). High-pressure storage of cabin air (70% nitrogen, 30% oxygen) is used for cabin repressurization and for nitrogen make-up due to leakage. The use of a mixed gas for cabin repressurization was deemed simpler and more reliable than separate nitrogen and oxygen sources.
Cabin ventilation and atmosphere revitalization is divided into 2 loops as shown in Figure 8.1. An outer loop provides cabin air recirculation, filtration, and temperature control. A top-down air flow is envisioned that minimizes contamination by fine lunar dust brought into the cabin on the spacesuits. An inner "suit loop" provides CO_2 removal, trace contaminant control, and humidity control during both normal cabin operations and during umbilical operations. The suit loop is purged with oxygen during the pre-EVA spacesuit purge and remains pressurized with 100% oxygen (at the spacesuit pressure) when the cabin is depressurized (while closed off from the outer cabin loop). The purged oxygen is vented overboard to prevent an elevated (> 30%) oxygen concentration inside the cabin.

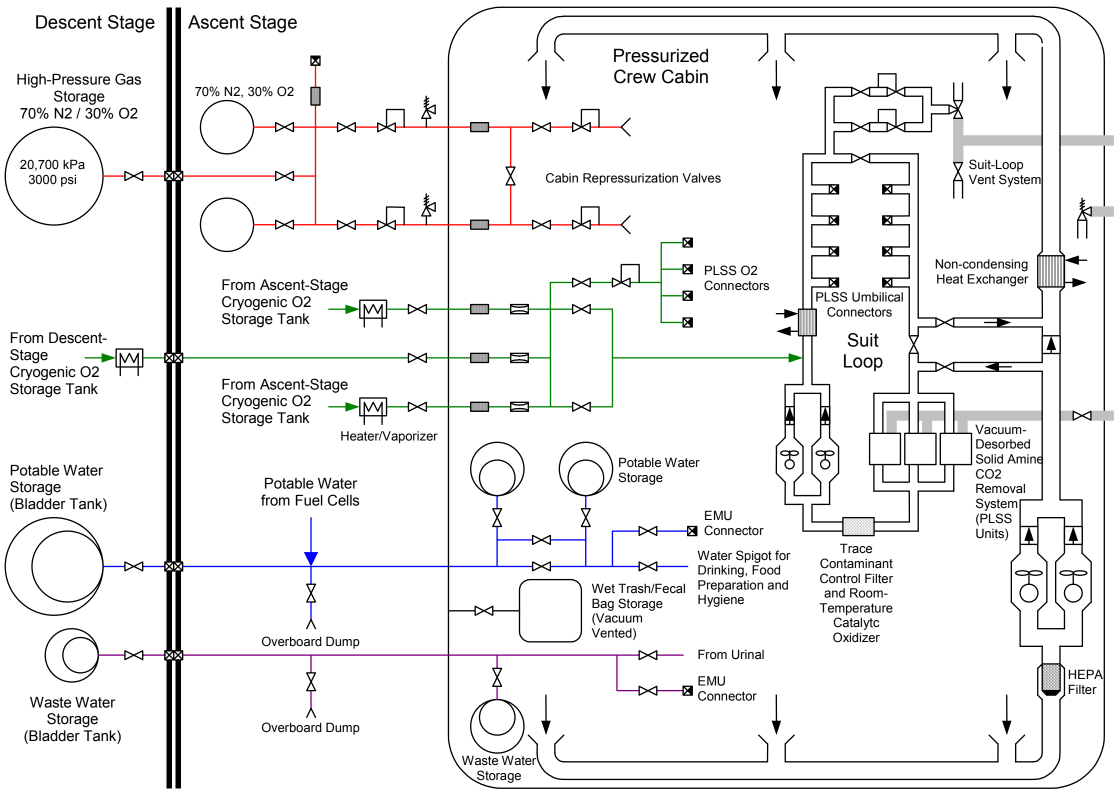


Figure 8.2: Simplified schematic of the L1 Lunar Lander ECLSS

Table 8.4: ECLSS Reference Design for the L1 Lunar Lander

Function/Subfunction	Technology
Atmosphere Control and Supply	
mixed-gas storage (70% N_2/30% O_2)	high pressure storage
oxygen storage	cryogenic storage[1]
atmosphere pressure control	software operated (X-38) + regulators[2]
atmosphere pressure monitoring	high accuracy (ISS) pressure sensors
Atmosphere Revitalization	
carbon dioxide removal	regenerative (vacuum desorbed) solid amine[3]
trace contaminant control	activated carbon + ambient-temperature catalytic oxidizer (Shuttle)
atmosphere composition monitoring	major component only (Shuttle) monitors
oxygen generation	n/a[4] (not justified for mission duration)
Temperature and Humidity Control	
cabin ventilation	ducting and blowers
atmosphere temperature control	non-condensing heat exchanger (accounted for in thermal control system)
atmosphere humidity control	function provided by carbon dioxide removal system
particulate and microbe control	high efficiency particulate air (HEPA) filters
Fire Detection and Suppression	
fire detection and suppression	smoke detector/halon
Water Recovery and Management	
potable water storage	bladder tanks
water microbial control	iodine microbial check valve
water quality monitoring	n/a (no water processing)
humidity condensate storage	n/a (all humidity removed by carbon dioxide removal system and vented outside)
hygiene wastewater storage	n/a (minimal hygiene water usage with use of disposable wet wipes; towels air dry)
urine/brine storage	bladder tanks
wastewater processing	n/a (not justified for mission duration)
Food Supply	
food supply	packaged Shuttle-type food system
food production	n/a (not justified for mission duration)
Waste Management	
urine collection	simplified Mir commode/urinal
feces collection and storage	simplified Mir commode/urinal

solid waste processing and storage	bag and store in solid waste storage bin (vacuum vented)

[1]Common cryogenic oxygen storage for propulsion, power, and ECLSS.
[2]Primary pressure control by regulators during cabin repressurization.
[3]Initial baseline assumes carbon dioxide removal units identical to that in the spacesuit potable life support system (PLSS) (3 units in parallel).
[4]n/a = not applicable (function not required for this design).

Potable water is obtained primarily from fuel cells and stored in bladder tanks. An initial storage of 40 kg of potable water was estimated to be required based on the expected water use/production profile.

A simplified commode/urinal system is included (similar to Mir commode) with charcoal-filtered air flow to remove odors. (Fecal handing and odor control were reported to be significant problems in the Apollo missions.) Additional crew accommodations are shown in Table 8.2. This list was derived from a general "crew accommodations" reference and should be reviewed further as the design matures. A crew health care system for the L1 Lunar Lander is described in Table 8.3. T.Sullivan/SD3 provided this list.Table 8.5: Lunar Lander Crew Accommodations[1]

Equipment/Supplies
Galley
spigot for food hydration and drinking water
cooking/eating supplies
Waste Collection System
waste collection system supplies
backup fecal/urine bags
Personal Hygiene
handwash/mouthwash spigot (may be same as galley spigot)
personal hygiene kit
hygiene supplies
Clothing
clothing (no laundry)
Housekeeping
vacuum cleaner
disposable wipes for housecleaning
trash bags
Operational Supplies and Restraints
operational supplies
restraints
Maintenance
tools
test equipment
Photography
equipment and supplies
Sleep Accommodations
sleep provisions (hammocks)

[1]List and sizing data obtained largely from Stilwell, D., R. Boutros, and J. H. Connolly, "Crew Accommodations", Chapter 18 *in Human Spaceflight: Mission Analysis and Design*, Draft, W. J. Larson and L. K. Pranke, editors, Space Technology Series, McGraw-Hill, 1999. Food and waste collection system (commode) shown under ECLSS above.

Table 8.6: Lunar Lander Crew Health Care[1]

Equipm pplies
Health Maintenance System (HMS)
crew contamination protection kit
advanced life support pack
ambulatory medical pack
crew medical restraint system
defibrillator
respiratory support pack
HMS ancillary support pack
emergency health care
medical equipment computer[2]
HMS consumables
Environmental Health System
tissue equivalent proportional counters
passive dosimetry
crew microbiology kit

[1]List and sizing data provided by Tom Sullivan/SD3.
[2]Medical computer capability is required on all vehicles, but can be a shared resource to avoid mass and volume impact.

8.4 Technology Needs and Design Challenges

The only advanced technology used in this primarily open-loop ECLSS design is the regenerative CO_2 removal system. While this basic technology has been tested in the Space Shuttle RCRS system, new solid-amine adsorbents have been proposed that have higher capacity. In addition, the small PLSS CO_2 removal unit requires much more frequent cycling than the RCRS, and lifetime issues for both the adsorbent and associated hardware (such as valves) require further investigation. A complete assessment of the relative capacities for both CO_2 and moisture removal as a function of atmosphere conditions is also required before the feasibility of this concept can be fully addressed.
The ability to provide spacesuit umbilical support with complete cabin depressurization from a mixed atmosphere is a design challenge. The concept presented here appears feasible, but requires further study.

9.0 L1 Lunar Lander Avionics System

9.1 Functional Description and Design Requirements

The Avionics system for the L1 Lunar Lander provides for the command, control, communications, and computation required for the carrying out the L1 Lunar Lander mission from launch to Lander disposal. These provisions reside in the context of human flight critical operations and, therefore, must meet the associated reliability requirements.

9.2 Trades Considered

No particular trades were considered for the initial design of the avionics system for the L1 Lunar Lander vehicle. The avionics system proposed for the Lander was based on similarity to other mission critical systems under development such as the X-38 or previous studies such as Human Lunar Return. This was considered adequate for the level of definition required for meeting the power and mass requirements for this particular phase of the design.

9.3 Reference design description

Figure 11.1 describes a high-level view of the avionics architecture. The heart of the avionics system is a set of flight computers which control all aspects of the flight including rendezvous and docking with the Solar Electric Propulsion Unit (SEP), powered flight to the Gateway at Lunar L1, rendezvous and docking with the Gateway, and finally the descent and ascent to and from the lunar surface respectively. In addition, the flight computers are responsible for overall system management and caution and warning information display.

An Inertial navigation system based on the ring laser gyro provides constant attitude information used by the flight computers in connection with either long range tracking or a lunar GPS to determine and maintain its inertial states and attitude knowledge. Initialization of the attitude is performed by a stellar attitude sensor automatically. A Ladar system is used to determine range required for the rendezvous operations and for fine range and relative attitude control for docking operations. During powered descent, a laser altimeter supplies accurate altitude for control of the descent trajectory and landing. A hazard avoidance system based on laser scanning assists the crew in avoiding hazards at the landing site or in redesignation of the landing site. The entire navigation system is capable of providing the information necessary for autonomous control of the lander for all operations.

S-Band communications systems will provide for the transmission of data, voice, and video directly to the earth and for reception of command data from the earth. A space-to-space radio system will support operations between the lander and the Gateway, the LTV, EVA crew members, and a rover or Habitat vehicle.

The video system will provide the obvious status views of various operational activities by the crew and of the surrounding environment whether in proximity to other vehicles or on the lunar surface. In addition, however, wide angle and stereoscopic cameras that comprise the video system support avoidance of hazards during the landing operation by providing visual imagery of the surrounding terrain when lighting conditions are appropriate. This information will be available to the crew via video cockpit displays.

Although the Lander is fully capable of autonomous control, it can also be manually controlled by the crew during all aspects of its operations. Crew input devices, crew displays, and caution and warning panels will provide the appropriate interfaces as required for manual control by the crew.

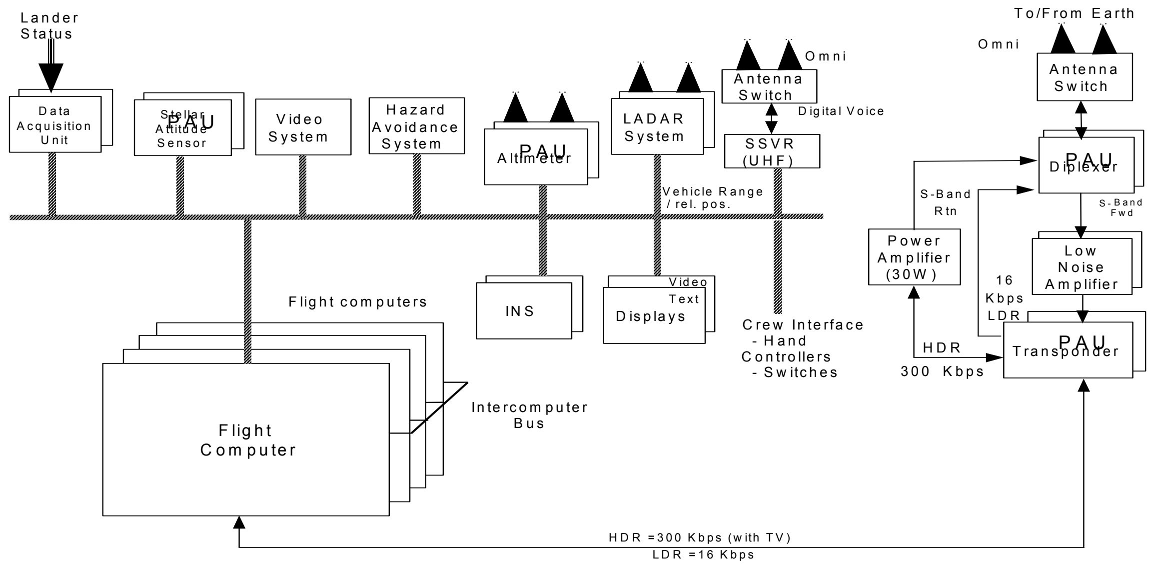


Figure 9.1 – L1 Lunar Lander Avionics Architecture

9.4 Technology needs and design challenges

Current technology of avionics systems was considered adequate for meeting the design guidelines of the Lander avionics systems. These guidelines focus on meeting launch mass constraints to which the avionics system, even with current technology, contributes only a small portion. It would certainly be expected that, by the time of implementation, advances in avionics technology would provide additional power, weight, size, and performance enhancements. The primary constraining factor to staying up to state-of-art technology is the requirement for radiation tolerance and electronic robustness of manned space flight critical systems.

10.0 Mission Success

10.1 Introduction

Safety, Reliability & Quality Assurance (SR&QA) involvement with the Exploration Office team has fostered the creation of an early safety and mission assurance plan for engineering concept studies that includes the following guidelines and products:

- The establishment of safety and reliability guidelines to assure crew and vehicle safety;
  - Design out hazards wherever possible;
  - Known hazards that cannot be eliminated by design will be reduced to an acceptable level by incorporating hazard controls into the system design;
  - When it is impossible to preclude the existence of known hazards, detection systems shall be used, to provide timely warning of the ensuing hazardous conditions;
  - Special operational procedures shall be developed to counter hazardous conditions when it is not possible to reduce the magnitude of an existing or potential hazard by design.
- The identification, tracking and documentation of hazards to crew and vehicle safety through a Preliminary Hazard Analysis;
- The establishment of preliminary Failure Modes and Effects Analysis (FMEA) for all subsystem and system configurations that will lead to the identification of a Critical Items List (CIL) for each design;
- The Performance of system reliability and availability analyses predicting logistics support levels and mission success probabilities using various software tools;
- The employment of qualitative and quantitative risk assessment methodologies for the management of risk to the program.

This safety and mission assurance plan for engineering concept studies has provided the engineering team with a more comprehensive understanding of the risk involved with conducting the planned missions. Furthermore, once a program is established, the plan will lead to better, and more defined mission requirements.

With the above plan outlined, SR&QA support has focused on three major areas that help make early safety and reliability, mission architecture and element design decisions. First, the daily presence of an SR&QA representative during the team meetings has helped provide safety and reliability insight to the subsystem engineers who incorporated these disciplines in the design and operations of the planned mission. Second, reliability/availability modeling of the subsystems that make up the elements within a planned mission has been completed to show the benefits of adding redundancy within sub-systems and to provide a best estimate of the maintenance and sparing requirements associated with the given mission architecture. Last, a Preliminary Hazard Analysis (PHA) was completed to identify aspects of the design that contain the most risk to the crew and vehicle. From these types of analyses, recommendations were incorporated into the mission architecture that positively affected hardware design and operating scenarios.

10.2 Scope

The scope of this report is to present results generated from above mentioned types of analyses for the L1 Lunar Lander. Quantitative results are based on Reliability Block Diagram (RBD) models derived in cooperation with the participating subsystem engineering leads. Qualitative safety analysis is based on a generic set of hazardous conditions typically encountered in human space flight and the design of the operations, elements, and systems that are part of the L1 Lunar Lander mission architecture.

10.3 Analysis Methodologies

10.3.1 Quantitative Analysis Methodologies

10.3.1.1 Development of RBD Subsystem Models

The RBD models produced to do the reliability and availability analyses are based on concepts generated for the L1 Lunar Lander by the L1 Lunar Lander team subsystem leads. The subsystem data captured in their concept templates and the corresponding block diagrams were used as a first cut at the subsystem architecture (RBDs). Once developed, the RBD models were then reviewed to close up any questions or issues and to gather more information about the repair-ability of the subsystems.

10.3.1.2 Quantitative Analysis Tools

Once a final RBD was established the model's failure probability was simulated using the Rapid Availability Prototyping for Testing Operational Readiness (RAPTOR) analysis software. This software randomly schedules failures using a Monte Carlo simulation of the system over the projected timeline and thereby predicts mission success, maintenance downtime and required spare parts.

10.3.1.3 Assumptions used in Quantitative Predictions

Some assumptions regarding input data must be made to do quantitative reliability and availability prediction analyses. The most important and consequential assumption is the failure rate data. To get a good set of failure rate data for the parts used in modeling the subsystems of the L1 Lunar Lander, historical data of similar systems in past and current NASA, commercial satellite and military programs are used. Also, the failure rate for all parts in the modeling completed for the L1 Lunar Lander is assumed to be constant, so, infant mortality and wear out are not included. Preventive maintenance or sparing for consumables, such as the vent loop filters is not included. Finally, Mean Time to Repair (MTTR) and repair-ability data were based on subsystem engineering experience. Thus, these values were assumed since it is application specific. As with any modeling activity, these assumptions contribute to a certain amount of uncertainty in the final results. While we cannot say the results presented are perfectly representative of real life, they can be used to provide a best estimate of system availability performance. This will enable the team to do trade studies and sensitivity analysis that will contribute positively to the element design.

10.3.2 Qualitative Analysis Methodologies

10.3.2.1 Preliminary Hazard Analysis

The purpose of the PHA is to identify safety-critical areas, to identify and evaluate hazards, and to identify the safety design and operations requirements needed in the program concept phase. The PHA provides management with knowledge of potential risks for alternative concepts during feasibility studies and program definition activities.

10.3.2.2 Failure Modes and Effects Analysis

In the process of conducting a FMEA, each hardware item is analyzed for each possible failure mode and for the "worst case" effect. The analyst begins with block diagrams that illustrate the operation and interrelationships of functional entities of a system and provides the ability to tracing failure mode effects through all hardware levels. In the preliminary phase of hardware design, a table documenting the failure modes and effects can provide management with knowledge of potential short falls in the hardware and operational design relationships.

10.4 L1 Lunar Lander Results

10.4.1 L1 Lunar Lander Subsystem Availability Results

Availability results for each of the studied subsystems (redundancy configurations developed by the subsystem leads) making up the L1 Lunar Lander are presented in Chart 1. As seen in this chart, the ECLSS, TCS and Avionics subsystems contribute to the unreliability of the Lander system as a whole. With spares and repair these systems can attain a high level of availability since all or the majority of parts composing the subsystems are repairable. Also note that the EPS, PROP, ACS, SUIT and SEP subsystems were assumed to have no repair capability for this element. It is understood that the EVA Suit will be repairable. However, for this short mission it did not contribute a great amount to the Lander unreliability so sparing analysis was not conducted.

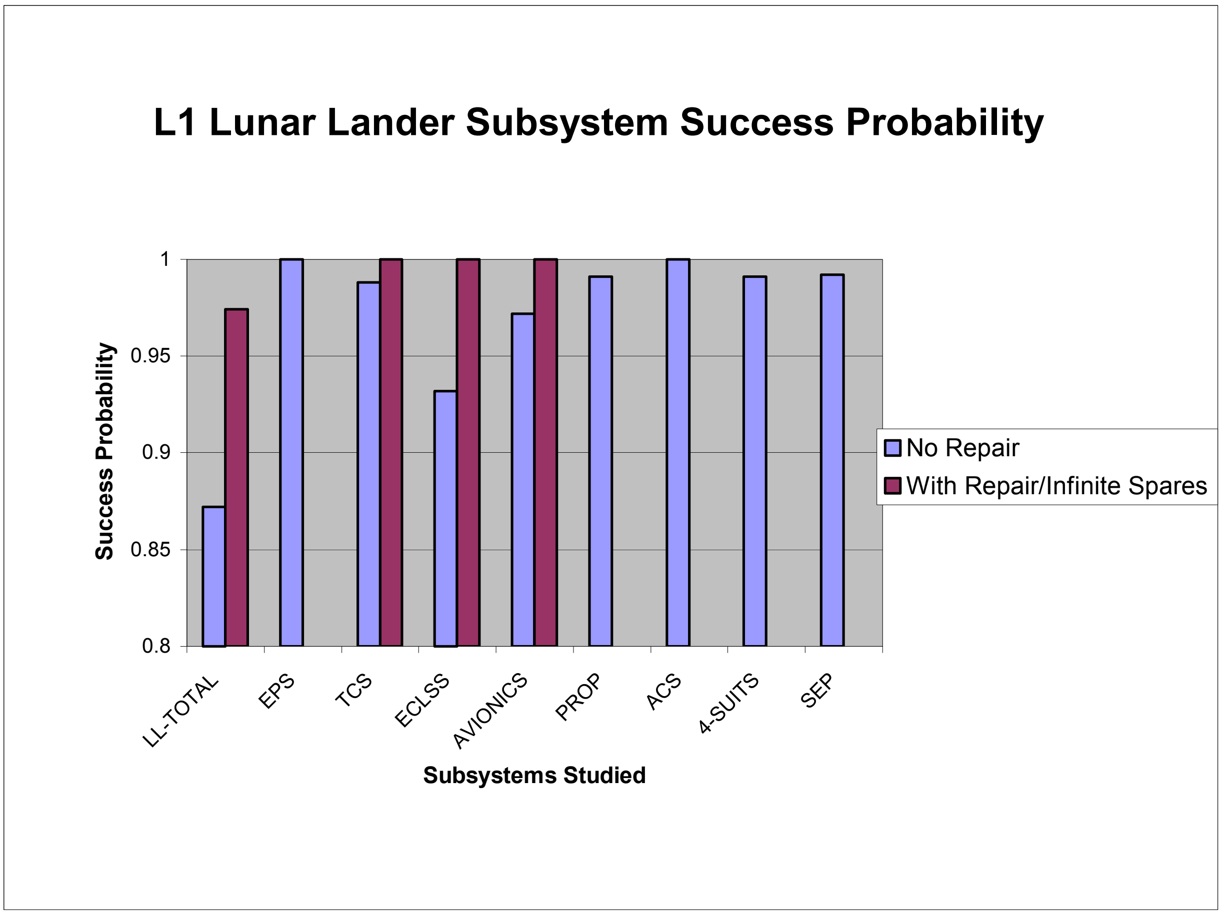


Figure 10.1. L1 Lunar Lander Subsystem Availability

Figures 10.2, 10.3 and 10.4 provide more detail about the sparing requirements for Avionics, TCS and ECLSS subsystems. These charts show the confidence of completing the L1 Lunar Lander mission per the established timeline against the spares allowed. For example, in Figure 10.2, if four spare Line Replaceable Units are brought along, there is a 98% confidence that the sparing needs of the avionics subsystem will be met. Likewise, there is 95% confidence that the avionics system will not need any spares. Also shown on these charts is the projected subsystem probability of success-this is the dashed line and is plotted against the secondary Y-axis. Listed in Table 10.1 is the sparing list for each of the subsystems listed in the charts. This list ranks in priority the spares that should be brought first through the maximum number listed for each subsystem. [It should be noted that there is a 99.99% probability that if all these spares are allowed, the Avionic, TCS and ECLSS subsystems could attain the availability, "with repair," listed in Figure 10.1.]

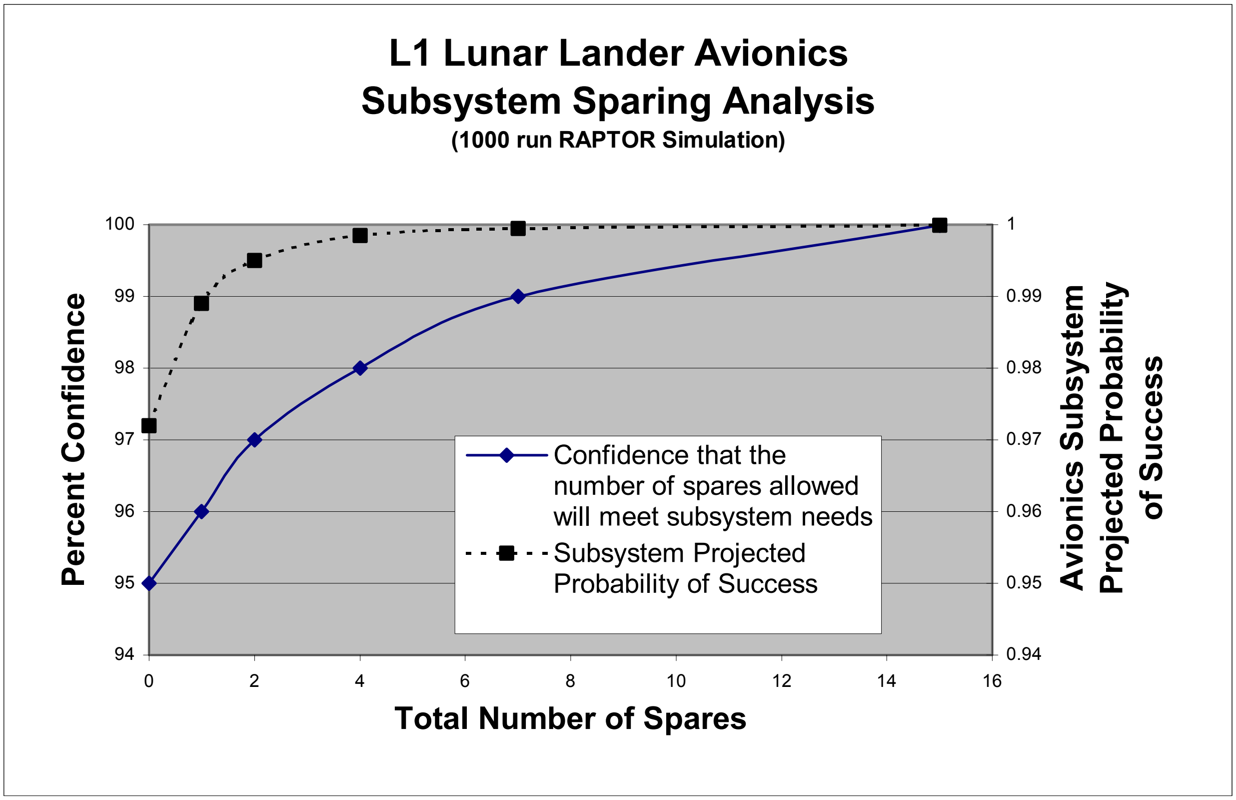


Figure 10.2. Avionics Subsystem Sparing Results

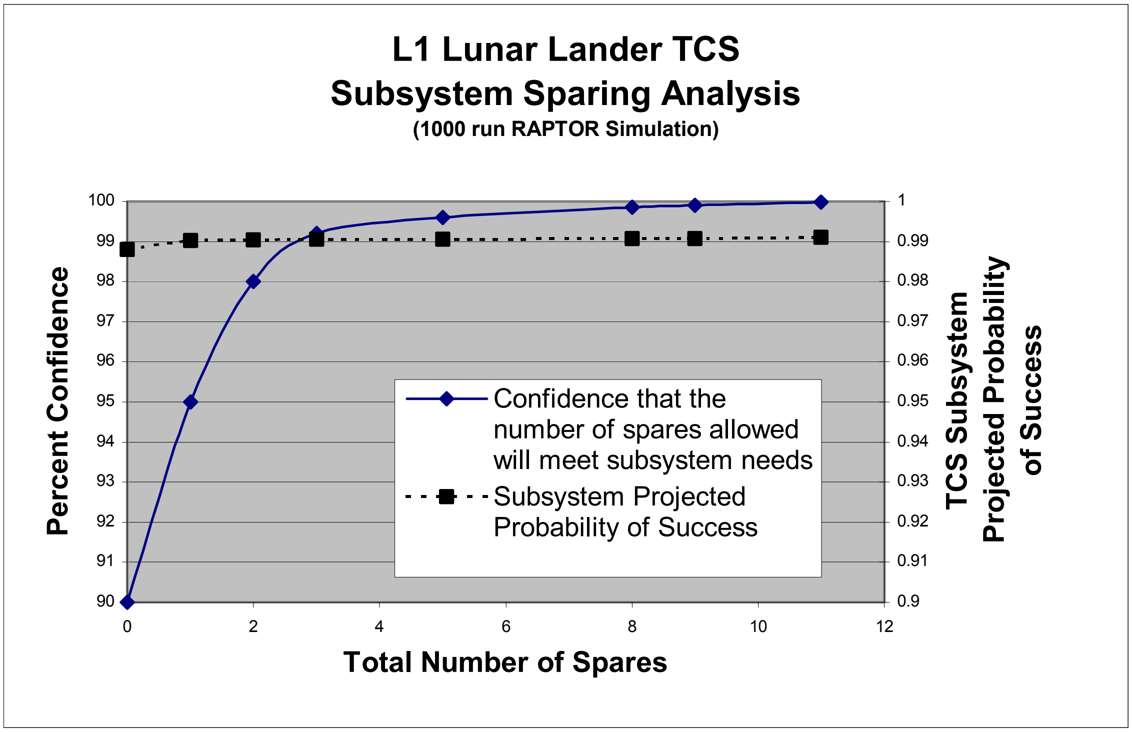


Figure 10.3 TCS Subsystem Sparing Results

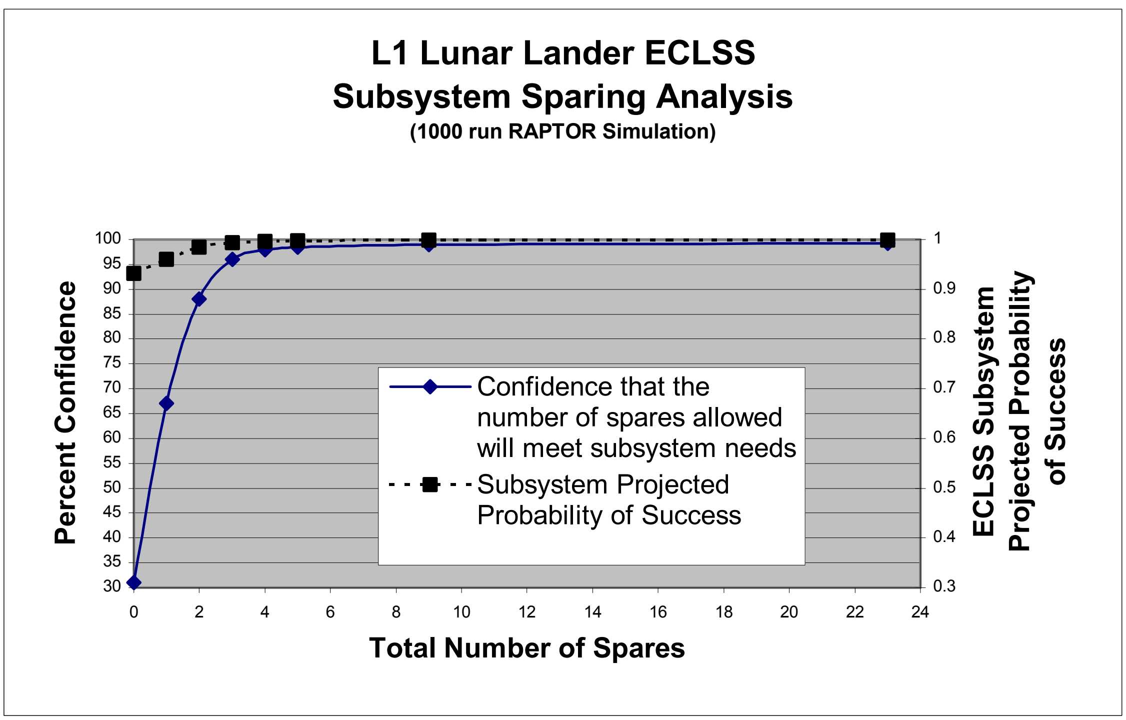


Figure 10. 4: ECLSS Subsystem Sparing Results

Table 10.1. L1 Lunar Lander Prioritized Cumulative LRU Sparing

Avionics	ECLSS	TCS
Antenna Switch	**Oxygen Partial Pressure Sensor**	**Pump**
Inertial Navigation System	**Oxygen Partial Pressure Sensor**	**Accumulator**
Central Computer	**Oxygen Partial Pressure Sensor**	**Flow Control Valve**
Central Computer	**Oxygen Partial Pressure Sensor**	**Gas Trap**
S-Band Transponder	**Blower**	**Pump**
Remoter Power Controller	**Water Delivery Valve**	**Temperature Sensor**
Video Tape Recorder	**2-Way Valve**	**Check Valve**
Modem	**Pressure Control Panel**	**Accumulator**
GPS Front-End Computer	**Smoke Detector**	**Gas Trap**
S-Band Antenna	**Cabin Purge Valve**	**Pressure Sensor**
Gimbaled Star Tracker	**Water Separator**	**Filter**
Antenna Switch	**EMU Battery Charge Unit**	
Data Acquisition Unit	**Positive Pressure Relief Valve**	
Video Monitor	**Total Pressure Sensor**	
High-Rate Antenna	**Water Delivery Valve**	
	Remote Power Controller	
	Air Mass Flow Meter	
	Fan Motor Controller	
	Oxygen Partial Pressure Sensor	
	Vent Fan	
	Total Pressure Sensor	
	Remote Power Controller	
	CO2 Amine Swing Bed	

10.4.2 L1 Lunar Lander Preliminary Hazard Analysis

Shown in Table 10.2 is the PHA for the manned phase of the L1 Lunar Lander. This analysis outlines both generic and unique operations hazardous conditions and lists their effects and controls. Controls listed reflect the current L1 Lunar Lander hardware design and operating scenarios as outlined by the concept design engineering team. Further real time interaction with this team is needed to provide more detail into the analysis.

Table 10.2 is on the following pages.

Table 2. L1 Lunar Lander PHA (manned phase)

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS
LL-01	**Contamination in habitable volume**	Lunar Dust	Respiratory irritation; Subsystem degradation	EVA post-ops procedures; Vacuum; ECLSS
		Leakage of TCS media	Respiratory irritation; Mucous membrane irritation	ECLSS; Adequate detection of leakage;
		Human by-products	Biological exposure to crew	Adequate crew procedures for isolation and containment
		Payloads/Science/ Lunar samples	Respiratory, Mucous membrane, skin irritation	Adequate crew procedures for isolation and containment of samples; adequate monitors
		Tool/Equipment Battery Leakage	Injury to crewmember	Battery design/containment
LL-02	**Electrical Shock**	Inadequate grounding	Injury or death to crewmember	Design; Testing; Redundancy
		Improper Circuit Design	Injury or death to crewmember	Proper sizing of electrical equipment and wire sizing so steady state currents do not exceed design
		Static Discharge	Injury or death to crewmember	Adequate measures for controlling potential
LL-03	**Environments**	Lunar Surface	Injury or death to crewmember	Accepted Risk; adequate crew training
		Thermal	Exceed lower or upper thermal limit of crew/vehicle components	Adequate and redundant TCS
		Acoustics	Physiological and psychological effects on crew	Adequate noise requirements and crew procedures
		Radiation	Long-term Crew Health; Carcinoma	Accepted Risk, minimum radiation protection by design; Adequate monitoring of solar activity

Table 2. L1 Lunar Lander PHA (manned phase)

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS
LL-04	**Fire/Explosion**	Flammable Materials	Loss of Crew/Vehicle	Material Selection, FDS
		Improper Circuit Design	Loss of Crew/Vehicle	Proper sizing of electrical equipment and wire sizing so steady state currents do not exceed design, FDS
		Ignition Sources	Loss of Crew/Vehicle	Preclude ignition sources by design, FDS
		High Pressure Vessel rupture	Loss of Crew/Vehicle	Material Selection; fracture controls (leak before burst); PPRV, FDS
		High concentration of Oxygen	Increased flammability of materials	Redundant O2 Partial Pressure sensing and control , Material selection, FDS
LL-05	**Impact/Collision**	Collision with Lunar Surface	Loss of Crew/Vehicle	Design for abort during descent; redundant altimeter and surface scanner
		MMOD	Loss of Crew/Vehicle	MMOD protection designed to shield Lander
		Inadequately restrained equipment	Loss of Crew/Vehicle	Adequate design of restraints; Adequate crew procedures for stowage of items
		Collision with LL1 Gateway	Loss of Crew/Vehicle	Redundant Lander and docking range finder; Single fault tolerant Reaction Control System; dual fault tolerant attitude sensing
		Loss of vehicle attitude control	Loss of Crew/Vehicle	Accepted Risk; zero fault tolerant for RCS control system
		Impact of Rotating or moving equipment	Injury or death to crewmember	Design and crew procedures.
LL-06	**Loss of Habitable Environment**	Depressurization	Loss of Crew/Vehicle	Adequate MMOD protection. Adequate design for pressurized volume with backup procedures to use EVA suit.
		Loss of O2 Supply	Loss of Crew/Vehicle	Redundant O2 Partial Pressure Supply, Sensing and Control with backup procedures to use EVA suit.

Table 2. L1 Lunar Lander PHA (manned phase)

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS
		Loss of CO2 removal capability	Loss of Crew/Vehicle	Redundant CO2 Removal capability with back up procedures to use EVA suit.
		Loss of TCS	Loss of Crew/Vehicle	Redundant loop TCS system
		Toxic Environment	Injury or death to crewmember	Materials Selection, Trace Contaminate Control System.
LL-07	**Physiological/Psychological**	Acceleration, shock, impact & Vibration	Injury or death to crewmember	Adequate design of restraints; Adequate crew procedures for stowage of items
		Effects of Pressure Changes on Crew	Possible Injury to crewmember	Adequate crew safety procedures for EVA pre-breath
		Illness/Incapacitation of Crew Member	Injury or death to crewmember	Crew Health equipment and procedures
		Excessive Noise	Possible Injury to crewmember	System designed for low noise generation. Hearing protection used in areas of high noise generation
		Sharp Edges/Pinch Points	Possible Injury to crewmember	Hardware designed where they will not pinch or snag the crew or their clothing. Exposed surfaces are smooth and free of burrs
		EVA Workloads & Fatigue	Possible Injury to crewmember	Crew procedures established to minimize crew fatigue
LL-08	**Radiation**	Solar Flare	Injury or death to crewmember	Accepted Risk, Adequate monitoring of solar activity, maximum radiation protection
		Non-Ionizing Radiation	Injury or death to crewmember	Minimize radiation emittance and maximize protection of components sensitive to EMI . Minimize use of Ionizing radiation sources by design
		Ionizing Radiation	Injury or death to crewmember	Minimize use of Ionizing radiation sources by design

Table 2. L1 Lunar Lander PHA (manned phase)

HAZARD NO.	CONDITION	CAUSE	EFFECT	CONTROLS
LL-09	**EVA Operations**	Inability to return to Crew Habitat	Injury or death to crewmember	Crew EVA rescue procedures in place to secure injured crew member back to habitat
		Crew Injury	Injury or death to crewmember	Crew CHeCs medical equipment
		Inability to Re-pressure cabin after EVA	Loss of Mission	Redundant Cabin pressure control system, Sensing and Control with backup procedures to use EVA suit for crew return to Gateway.
		Contamination of crewmember from leaking RCS/Engine Thruster	Possible Injury to crewmember	Isolation valves for Upstream manifold and tanks.
LL-10	**Docking Operations**	Inability to Dock with Gateway	Loss of Crew/Vehicle	Backup procedures to use EVA suit to return to Gateway
		Inability to equalize pressure with Gateway	Loss of Crew/Vehicle	Backup procedures to de-pressure Lander and use EVA suit to return to Gateway

10.4.3 L1 Lunar Lander Preliminary Failure Modes and Effects Analysis

Reserved for future analysis of the subsystems for the subject element.

10.5 Findings

There is a good chance that zero spares will be needed for the subsystem configurations studied. However, having selected spares along for the Avionics, TCS and ECLSS subsystems would increase the subsystems probability of success as shown in Charts 2, 3 and 4. The ultimate recommendation would be to increase the base reliability of these and the other subsystems that make up the L1 Lunar Lander so maintenance demand on the short crew stay in this element could be minimized. Further iterations on these concept subsystems are needed to provide insight into the redundancy levels needed to achieve a high rate of mission success.

10.6 Conclusions

For this short mission profile safety requirements will dominate subsystem design. Certain levels of redundancy in critical systems are required for human rating a space vehicle. High levels of reliability could probably be achieved with such a short mission without meeting these types of safety requirements. In that light, the Preliminary Hazard Analysis, which evaluates safety of the design architecture, would be the first source for the development of Safety and Mission Assurance design requirements for the ensuing program.